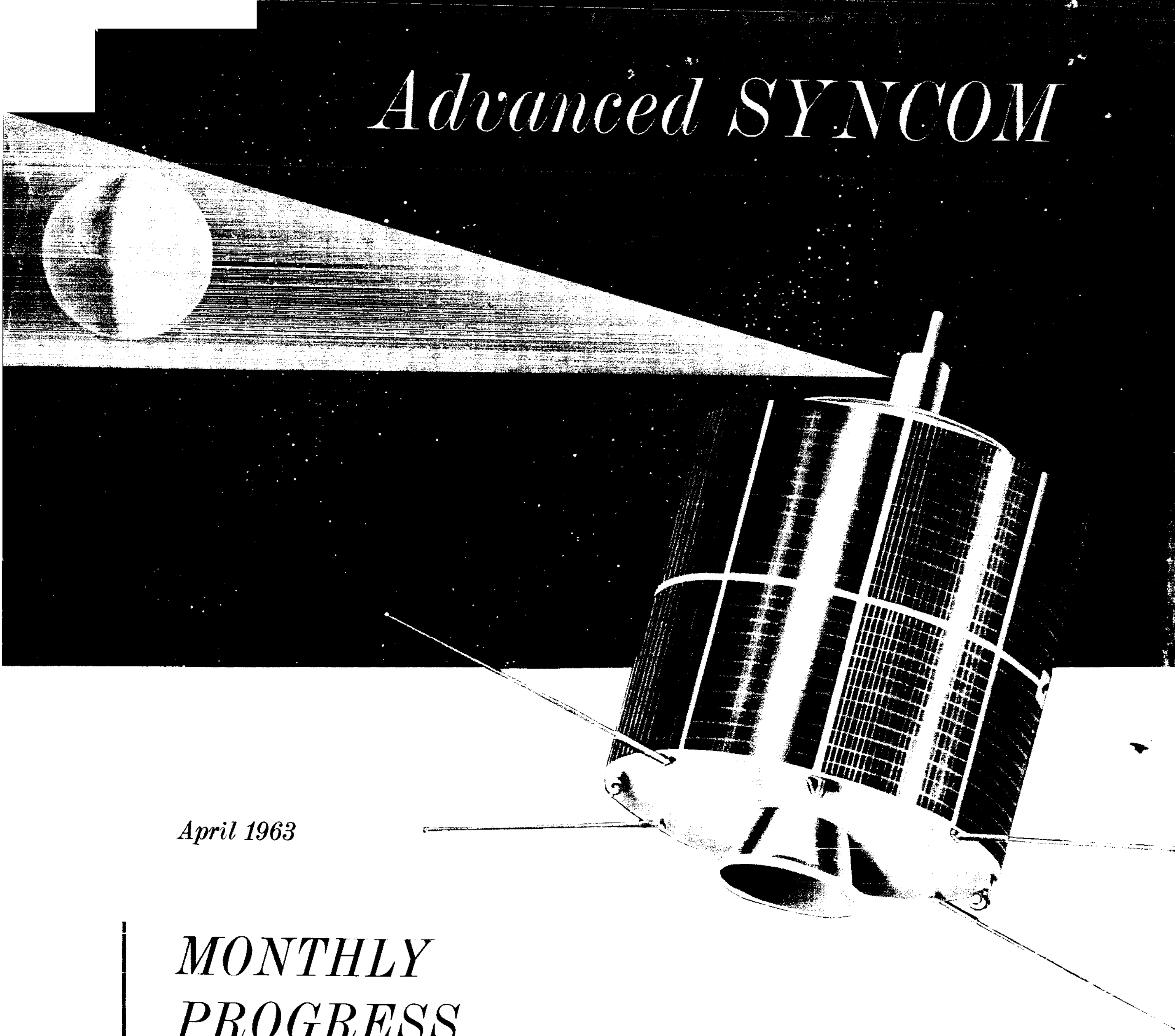
Advanced SYNCOM



April 1963

MONTHLY PROGRESS REPORT

Supplement to Summary Report

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NASA Contract 5-2797
SSD 3290R

AEROSPACE GROUP
SPACE SYSTEMS DIVISION
HUGHES AIRCRAFT COMPANY
CULVER CITY, CALIFORNIA

HUGHES

HUGHES AIRCRAFT COMPANY

AEROSPACE GROUP

SPACE SYSTEMS DIVISION

EL SEGUNDO, CALIFORNIA

15 May 1963

SUBJECT: Advanced Syncom Monthly Progress Report for April 1963 (Supplement 1 to Summary Report dated 31 March 1963)

TO: Mr. Robert J. Darcey
Program Manager, Syncom
Goddard Space Flight Center
Code 621
Greenbelt, Maryland

Attached are copies of the Advanced Syncom Monthly Progress Report for April 1963. This report, in addition to supplying April progress information, provides information supplemental to the Summary Report dated 31 March 1963.

The engineering model structure (T-1) was dummied to launch weight and instrumented to obtain structural design and dynamic response data during a series of environmental tests. The initial response surveys were completed and a preliminary analysis of the data is included in the attached report.

The results of an enumerative discussion comparing possible advantages and tradeoffs of a feasible multiple-axis stabilized system design with the present Syncom II spin-stabilized design are included. The study indicates the relative ease of incorporating meaningful redundancy in a spin-stabilized design as contrasted to the multiple-axis design. Further, the approach employed in closing the control loop through ground control stations results in a simplified attitude control system design in terms of sensors and control components.

Several additional spacecraft and ground support equipment system engineering documents were completed. A definition of anticipated spacecraft system tests, test criteria, and block diagrams of the spacecraft and ground support equipment required is provided.

HUGHES AIRCRAFT COMPANY

Paul E. Norsell

P. E. Norsell
Manager, Systems Development
Advanced Syncom

Advanced SYNCOM

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CONTENTS

1. INTRODUCTION

The use of communication satellites has been recognized to answer the need for greatly expanded global communications capability. It has been a major effort of the United States Government and of industry to develop a satellite relay system at the earliest possible time.

Under NASA Goddard Space Flight Center Contract NAS-5-1560, Hughes Aircraft Company developed the Syncom I spacecraft to be orbited by NASA Delta launch vehicles and used in conjunction with Department of Defense Advent ground stations for the performance of inclined synchronous-orbit communication experiments during 1963.

The Syncom I spacecraft will demonstrate a simple spin-stabilized design capable of being placed in a synchronous orbit. At the same time, it will be demonstrated that a simple pulse-jet control system can provide the stationkeeping necessary to maintain a synchronous orbit.

Additional important mission objectives of the NASA communication satellite program include the demonstration of a "stationary" or equatorial, synchronous orbit, conduct of system orbital life tests, demonstration of new wide-band services on a transoceanic basis, and demonstration of a system accessible to all nations.

Under NASA Goddard Space Flight Center Contract NAS-5-2797, Hughes is conducting feasibility studies and advanced technological development for an advanced, stationary, active repeater communication satellite. A Summary Report covered the technical progress achieved during the original contract period and details the system configuration resulting from the system studies. This supplementary report covers further studies which have been made under modification two to the above contract and the accompanying technical direction.

2. SYSTEM DESCRIPTION SUMMARY

The global communication system based on the Advanced Syncom stationary active repeater communication satellites will be compatible with all current types of common carrier traffic, typified by voice communications, teletype, and monochrome and color television signals. The system can provide service quality consistent with CCIR standards. Numerous ground stations can be readily accommodated, with each station able to communicate with any or all other stations at any time.

The voice communication capacity of each of the satellite transponders is 600 two-way telephone conversations, with ample margin over CCIR standards, which can be realized by using fixed, nontracking, 85-foot-dish antennas. Each satellite contains four such transponders, providing a total system capacity through the satellite of 2400 two-way voice channels. Alternately, the system can accommodate television or other wide-bandwidth signals through any of the transponders, again with ample margin over available CCIR standards.

The spin-stabilized satellite is launched by the Atlas-Agena D launch vehicle in conjunction with a third-stage apogee injection rocket carried integrally within the spacecraft. Bipropellant rocket reaction jet control systems provide thrust to correct anticipated initial errors in orbit parameters due to launch vehicle guidance tolerances. These bipropellant systems are also used to orient the spin axis of the satellite perpendicular to the orbital (equatorial) plane, and to correct periodically the parameters of the orbit to maintain the satellite stationary to within 0.1 degree throughout the satellite life.

The spinning satellite contains a phased-array transmitting antenna with electronic controls to maintain its highly directional pencil-beam pattern directed toward the earth. Four independent dual-mode communication transponders with efficient traveling-wave tube final power amplifiers provide alternate modes of operation corresponding to the type of communication to be repeated.

Solar cells provide 135 watts of electrical power, which allows a margin over the requirements for continuous, simultaneous operation of all equipment and battery charging circuits.

The satellite, exclusive of apogee motor, weighs 600 pounds when fully loaded with reaction jet control system propellants. The apogee motor and control system tankage are sized to accommodate the maximum payload capability of the Atlas-Agena D for this mission, a satellite weight of 650 pounds, exclusive of apogee motor, fully loaded with control system bipropellants. Apogee motor propellant is off-loaded to the requirements of the less than maximum weight satellite configurations.

3. COMMUNICATION SYSTEM DESIGN

Frequency assignments have been made for four channels as shown in Table 3-1. These frequencies satisfy the following relationships:

$$f_{in} = \frac{193}{128} f_{out}$$

$$f_{beacon} = f_{out} \left[1 + \frac{61}{(128)^2} \right]$$

TABLE 3-1. CHANNEL FREQUENCY ASSIGNMENTS

Channel Number	Input Frequency, (Ground to Spacecraft)	Output Frequency, (Spacecraft to Ground)	Beacon Frequency, (Spacecraft to Ground)
1	6019.325	3992.09	4006.95
2	6108.275	4051.08	4066.16
3	6212.10	4119.94	4135.28
4	6301.05	4178.93	4194.49

4. ADVANCED TECHNOLOGICAL DEVELOPMENT PROGRAM

PROGRAM SUMMARY

The Advanced Technological Development Program for an advanced, stationary, active repeater communication satellite includes research and development through fabrication and demonstration of engineering models of a multi-element phased array transmitting antenna and associated control circuits, a collinear array receiving antenna, a dual-mode communication transponder incorporating a traveling-wave tube final power amplifier, a spacecraft structure, and a hot gas reaction control system. Also included were studies of system design feasibility, preparation of performance and test specifications, demonstration planning, and conduct of preliminary engineering acceptance demonstrations.

Initial results of the system design feasibility studies were reported in "Initial Project Development Plan" in August 1962. The studies were continued in parallel with the advanced development work throughout the contract period and, for the period through 31 March, were reported in "Syncom II Summary Report."

On 4 April 1963 NASA issued Modification Two to contract NAS-5-2797 and a Technical Direction Order which clarified reporting requirements for this supplemental report. Work has continued throughout the period in most of the areas reported on in the Summary Report. The technical effort on the program was completed on 28 April with the preparation of the material contained in this report. Specific objectives which have been accomplished are given in the following paragraphs.

An analytical report on the comparison of a three-axis versus a spin-stabilized with de-spun antenna communication satellite has been completed. An advanced bill of materials and advanced preferred parts, materials, and processes lists have been compiled. A reliability failure mode analysis plan and a quality control operating plan have been generated.

System engineering has continued with the issuance of preliminary interface specifications on RF and electrical, and mechanical interfaces; preliminary spacecraft subsystem performance requirements; a system test document; a design criterion for support transponders; and block diagrams of both the spacecraft and the ground support equipment.

Effort on the various subsystems has proceeded. Further definition of the telemetry and command system has been accomplished. A power supply design specification has been issued. Test plans for the following critical components have been prepared: sun sensors, central timer, batteries, and separation switches. A simplified transponder has been designed which eliminates the need for two master oscillators in the multiple-access transponder and reduces the spread of IF frequencies required for both transponders. Engineering drawings of the stripline design for the communication transmitting antenna have been completed. The preliminary specification for the central timer has been issued.

Environmental testing of the T-1 structure is under way and a preliminary response survey has been completed in the thrust axis and one transverse axis. Engineering data have been obtained on various transponder minor control items as a function of temperature and power levels. Radiation patterns have been measured on the phased array antenna. Construction of six additional traveling-wave tubes is under way. A battery charge regulator has been breadboarded and tested. Spin-rate control mechanism tests have continued.

Spacecraft weight summary reports have been up-dated. A preliminary review of the interface between the structure and the wiring harness has been documented. Redesign considerations for the mobile assembly fixture are continuing.

Subcontract direction has continued with the Marquardt Corporation on the bipropellant system and liaison with JPL on the apogee motor has been maintained.

5. ADDITIONAL CONFIGURATION AND SENSOR STUDIES

MULTIAXIS VERSUS SPIN STABILIZATION FOR SYNCOM ATTITUDE AND STATIONKEEPING CONTROL

Summary

This section presents a brief description of the functional sensing, processing, and control elements required for the attitude and stationkeeping control of a 765-pound spinning and nonspinning spacecraft designed for a Syncom mission of 3 to 5 years duration. Comparable procedural, performance, and reliability requirements imposed on key elements of the two system design concepts during the ascent, apogee boost, reorientation, and stationkeeping phases of the mission are listed; emphasis is placed on the reliability growth comparison of the two system designs resulting from the semiquantitative physical arguments developed. It is concluded that, although both design concepts can be implemented within the present state-of-the art component limitations, the inherently long lifetime desired of this spacecraft favors the relatively simple spacecraft hardware design of a spin-stabilized attitude control and stationkeeping system, where much of the attitude and position error data processing can be done at a ground control station with continuous visibility to the spacecraft.

The gyroscopic stabilization of a spinning spacecraft requires a very small equivalent closed-loop bandwidth for both attitude and stationkeeping control during and between periods of vernier jet thrusting as well as during apogee motor boost; thus the loop can be efficiently closed through a ground control station at a low equivalent sample data rate. This is not the case with a multiaxis control system, in which continuous control of attitude angles and rates must be available to maintain one body axis along the local vertical and a solar array face normal toward the sun.

Introduction

The main purpose of this discussion is to briefly describe the salient design features of a multiaxis attitude and stationkeeping control system and compare them with those of a spin-stabilized system to point out the relative complexity of the resultant system state-of-the-art components and control logic needed to meet the requirements of a Syncom mission. The results of the comparative discussion will be used to indicate the superior reliability growth potential of a spin-stabilized system because of its simple design and minimum number of moving components, which allows the incorporation of more meaningful redundancy at the subsystem level.

No attempt will be made to compare the relative weight, power, and volume of the control systems since this would require (and is quite sensitive to) rather detailed design knowledge of each complete system as well as a set of ground rules indicating what fraction of the above parameters should be allocated to the attitude and velocity control functions. (Any well-integrated system will have at least some of its components used in more than one function.) A rough comparison of total power will be attempted by scaling from existing system designs.

Any multifaceted comparison of this scope will inevitably involve unproven judgments, simplifications, and opinions despite attempts to justify critical arguments via computation. Thus, statements concerning typical design criteria for the multiaxis system will be taken from a knowledge of existing state-of-the-art designs whose performance (with respect to control of attitude and velocity) is comparable with that of Syncom, i.e., Orbiting Geophysical Observatory (OGO) and ADVENT. Some justification will be given for choosing the reaction wheel-gas jet design similar to that of OGO and ADVENT (as opposed to an all-jet system) based on limit cycle, fuel consumption, and thrust level arguments. Furthermore, the conclusions are based on the following (perhaps unnecessarily restrictive) assumptions concerning 1963 - 1964 state-of-the-art booster availability and spacecraft performance.

1) The launch vehicle is the Atlas D/Agena D combination capable of injecting about 1520 pounds into a transfer ellipse with an apogee radius equal to the synchronous radius, 22,752.5 nautical miles, an inclination of about 29 degrees (AMR launch plus range safety), and a period of about 10.5 hours.

2) An apogee boost velocity increment of about 6100 fps is imparted to the spacecraft to remove the transfer orbit inclination and circularize the transfer ellipse into a nominally synchronous (24-hour) orbit.

3) The nominal spacecraft weight (including apogee motor case of $\sim$ 100 pounds) at apogee motor burnout is 765 pounds.

4) The design lifetime of the Syncom mission is 5 years.

5) The spacecraft control system must have the ability to

   a) Maintain thrust attitude during boost *

   b) Remove final injection dispersions of inclination, period, and eccentricity

   c) Achieve and maintain a selected longitude over the equator with an error of less than ± 0.05 degree for the satellite lifetime

   d) Continuously point the communication antenna beam center along the local vertical with an error of less than ± 2 degrees with minimum interruption (near minimum traffic hours if necessary)

6) Redundancy is to be used wherever component operation is critical within the constraints of payload limitations because of the unprecedented long design lifetime required of a spacecraft with this complex function.

The approach is enumerative -- that is, verbal -- and block diagram descriptions of the two system configurations will be given and plausibility arguments using physical reasoning advanced in the comparative discussions of the key components of each system.

Multiaxis Configuration

The main advantage of a multiaxis stabilized configuration for a Syncom mission lies in its ability to direct a high gain transponder beam toward the earth with a simple reflector-type antenna that is body-fixed. With this in mind one would like to choose the simplest configuration with a minimum number of sensing and control components that would meet the performance requirements and payload constraints assumed above using state-of-the-art techniques.

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*Although this function and the apogee boost function itself may be accomplished with the Agena vehicle using a third burn, the loss of payload due to the staging principle makes this approach noncompetitive with Syncom II.

Choice of Control Components

Since the initial orbital and subsequent stationkeeping velocity correction requirements call for an incremental velocity capability of 1100 to 1200 fps for the 5-year period (Reference 5-1, section 5), gas jets with a reasonably high specific impulse fluid are required to minimize the propellant weight. A hot gas system with a specific impulse, $I_{sp} = 260$ seconds, will require about 95 pounds of propellant only (fuel plus oxider) to impart 1200 fps to an average spacecraft weight of 700 pounds (with no redundancy). Thus, with the knowledge that the additional propellant requirement for attitude acquisition and control is much smaller than the velocity correction requirement, one is tempted to use these hot gas jet nozzles for attitude control as well. However, with the allowable deadband, $\Delta\theta \simeq 2$ degrees per axis, it will be shown below that the low jet thrust level F needed to make the subsequent limit-cycle fuel consumption tolerable is beyond the present hot gas jet state of the art. In particular, the total impulse, I_T, required to accommodate the limit-cycle motion about each axis is estimated by

$$I_T = \frac{T\,(nF)^2\,\ell\,(\Delta t)^2_{min}}{I_x\,(\Delta\theta)} = I_{sp}\,W_p \tag{5-1}$$

where

T = mission time, seconds

$\cong 15.8 \times 10^7$ seconds (5 years)

n = number of jets controlling the x axis

$= 2$

ℓ = moment arm of jets to cg, feet

$\simeq 2$ feet

$(\Delta T)_{min}$ = minimum <u>on</u> time of jet, seconds

$\geqslant 0.01$ second; $(\Delta t)^2_{min} = 10^{-4}\ \text{sec}^2$

F = thrust of each jet, pounds

I_x = moment of inertia about control axis

$\approx$ 60 slug-ft^2 (mass distribution comparable to that of Syncom II)

$\Delta\theta$ = allowed deadband, radians

$\approx 35 \times 10^{-3}$ radian (2 degrees)

I_{sp} = specific impulse, seconds

$\leq$ 270 seconds

W_p = propellant weight, pounds

Thus, from Equation 5-1

$$W_p \geqslant \frac{(15.8 \times 10^7)(4)\, F^2 (2)(10^{-4})}{(60)(35 \times 10^{-3})(270)} = 223 \left[lb^{-1} \right] F^2 \qquad (5\text{-}2)$$

Equation 5-2 implies $F \leq 0.1$ pound in order to make W_p small compared with the equivalent weight of a reaction wheel control assembly about this axis (2.5 to 5 pounds). Hot gas jets with thrust levels lower than about 1 pound are not available (and probably will not be for some time due to nozzle throat design problems at low thrust levels). If a dual jet system were considered -- a hot gas system for velocity control and a cold gas, low thrust system (e.g., nitrogen, I_{sp} = 70 seconds) for attitude control similar to the one on OGO (section 6, Reference 5-2; OGO uses argon gas), then the more realistic value of $(\Delta t)_{min} \approx$ 30 milliseconds would still prove troublesome. The expression for W_p (using N_2 and Δt_{min} = 30 milliseconds) in Equation 5-2 becomes

$$W_{p_{N_2}} \approx W_{p_{HG}} \frac{(9)\,(270)}{70} = (223)\,(34.7) = 7750 \left[lb^{-1} \right] F^2 \qquad (5\text{-}3)$$

Hence, using a reasonable value of $F = 0.05$ pound to complete the sun acquisition mode in 5 to 10 minutes, for example, results in a cold gas propellant weight per axis of 17.8 pounds. Additional propellant weight will be needed to overcome cyclical torques (e.g., solar paddle rewind once per day, orbital correction thrust orientation) as well as the smaller secular torques (radiation pressure unbalance, magnetic field effects). The above arguments indicate that the practical alternative (from a weight viewpoint) to the low thrust cold gas system is a three-axis reaction wheel system similar to that of OGO (Reference 5-2) with sufficient momentum storage to accommodate the cyclical torques plus some temporary storage for secular torques so that desaturation gas jet firings may be chosen at an arbitrary time in one orbital period. The deadband per axis of the reaction wheel system should be large enough to avoid continuous operation of wheel motors and sufficiently smaller than the jet system deadband to allow rapid system convergence from a gas jet firing and thus supress its limit-cycling tendency. In addition, the hysteresis of reaction wheel switching function must be large enough (but smaller than the deadband) to avoid excessive operation of the motor as a result of sensor noise. A reasonable reaction wheel deadband value is 1 degree, about half that of the jet system (2 degrees), but detailed tradeoff studies and extensive simulations are necessary to arrive at proper design values for each axis. Similar statements apply to the design of the reaction-wheel size, weight, and saturation speed.

Having arrived at the selection of a wheel-jet control system, one would like to minimize the number of each component needed to effectively maintain three-axis body control. Figure 5-1 is a block diagram of a wheel-jet system (with first-order control decoupling of each axis) showing 12 jets and three reaction wheel assemblies. This modified OGO-ADVENT design uses jets 1 through 6 exclusively for attitude control; jets 7, 9, 10, and 12 are shared for yaw axis control (including de-spin) and orbital inclination removal; and jets 8 and 11 are used exclusively for in-plane control (longitude) of the orbit. Although actuation of the pitch jets (2 and 5) introduces some translational acceleration, this is not considered serious to warrant two more nozzles. (Similar arguments may be used to remove jets 4 and 6 by placing jets 1 and 3 to positions 1' and 3' in Figure 5-1 if a net weight saving results.) The thrust direction of jets 2 and 5 (also 1' and 3') may be canted inward (toward the yaw axis) to take advantage of the increased moment arm to the cg. To be sure, the application of some ingenuity can further reduce the number of jets but at the expense of some cross-coupling logic and more subtle rearrangement of the location of the principal axes of inertia than assumed here (along pitch, roll, and yaw axis). It will be shown later that the mode logic is sufficiently involved as it is (when compared to a spin-stabilized system) and any design consideration that would tend to complicate the logic is to be avoided without a detailed tradeoff study.

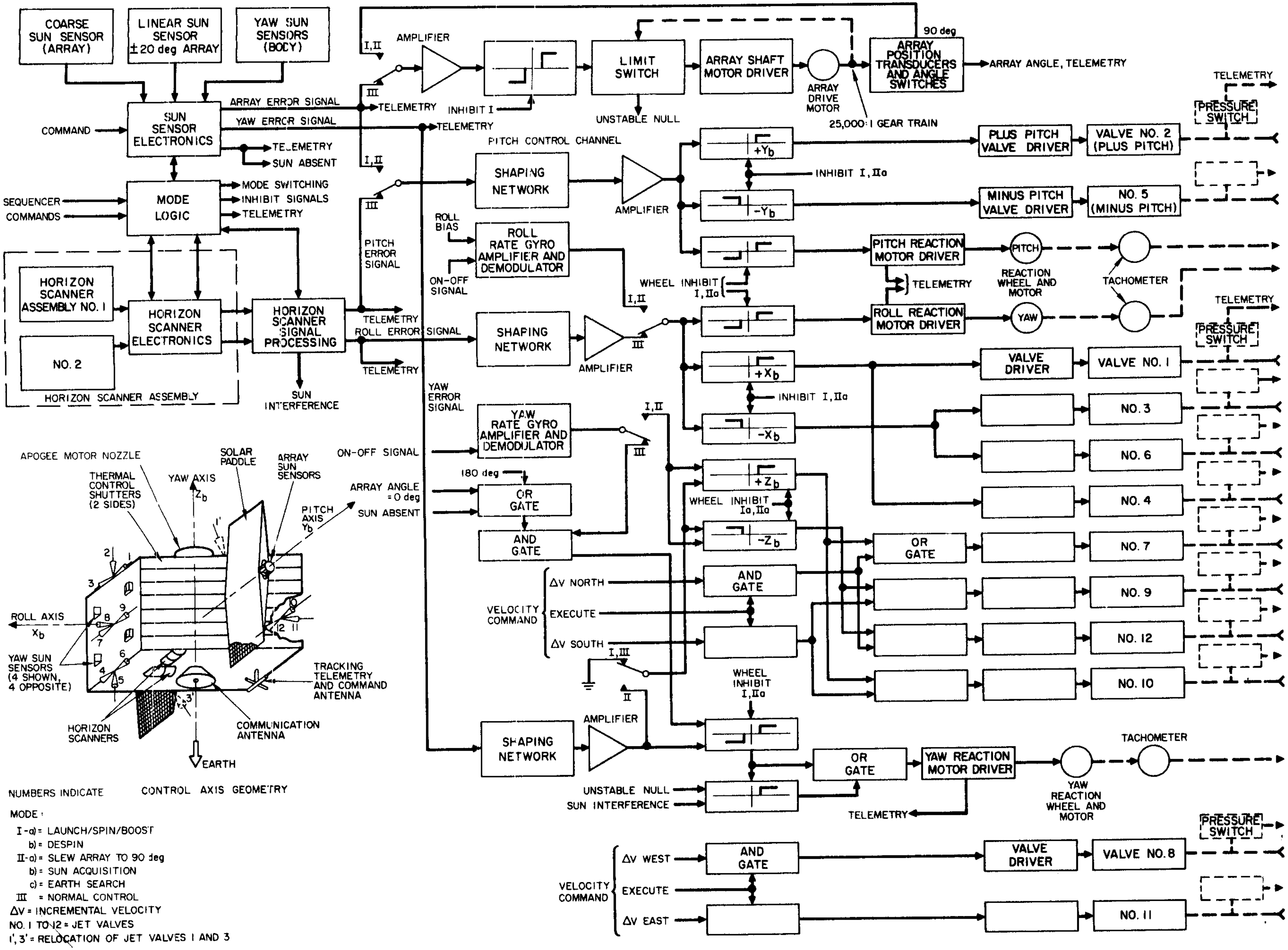


Figure 5-1. Multiple-Axis Altitude and Velocity Control System for Syncom with Single-Axis Solar Paddles

Solar Paddles Versus Body-Fixed Solar Array and Thermal Control

With one body axis (yaw axis in this design) constrained to point along the local vertical as the spacecraft moves in an equatorial orbit, one would like to choose a solar cell area configuration that would be simple and not conflict with other design constraints. Neglecting, for the moment, the effect of the inclination of the ecliptic to the equatorial plane and constraining the roll axis, say (via sun sensors), to remain pointed in the plane of the ecliptic (equatorial plane for zero inclination of ecliptic), then a body-fixed array would require solar cells covering all faces of the spacecraft that are parallel to the pitch axis in order to get adequate solar cell illumination during the 24-hour period, including the earth-pointing face containing the horizon scanner. Now, to maintain the illuminated cell area almost constant during the orbit, the communication and telemetry and command antennas should then be mounted on one of the end planes normal to the pitch axis, as in Figure 5-2. This would leave only one of the end planes for thermal control shutters. A configuration of this type would probably employ cylindrical symmetry as in the present Syncom design. The resulting reduced solar power efficiency is comparable to the spin-stabilized system but the thermal control problem is worsened since no temperature averaging due to spinning is available and only one end plane is available for thermal shutter control. Whether this configuration is adequate from a temperature control viewpoint will not be known until a heat balance study is made (assuming that the total system power requirements are comparable to those of Syncom II so that the inefficient use of solar cell area is also comparable, probably a dubious assumption in the light of some published ADVENT power requirements of over 400 watts excluding battery charge requirements as opposed to 125 watts for Syncom II including battery charge).

With the above reservations plus the observation that similar multiaxis systems such as OGO and ADVENT employ sun tracking solar paddles and use the two faces (of a rectangular parallelepiped) on the spacecraft that are normal to the solar paddle axis (pitch axis) for thermal shutter control, the design considered here will also use solar paddles whose axis is along the body pitch axis. Sun sensors mounted on the body faces normal to the roll axis will be used to constrain the body roll-yaw plane to remain in the ecliptic plane (yaw motion) except for noon, midnight, and eclipse conditions) while the sun sensors mounted on the paddle axis will be used to maintain the paddle face normal to the sun line (except for noon, midnight, and eclipse conditions). Thus the multiaxis system to be compared with a spin-stabilized design will have three-axis body control plus control of a single axis solar paddle relative to the spacecraft body (Figure 5-1).

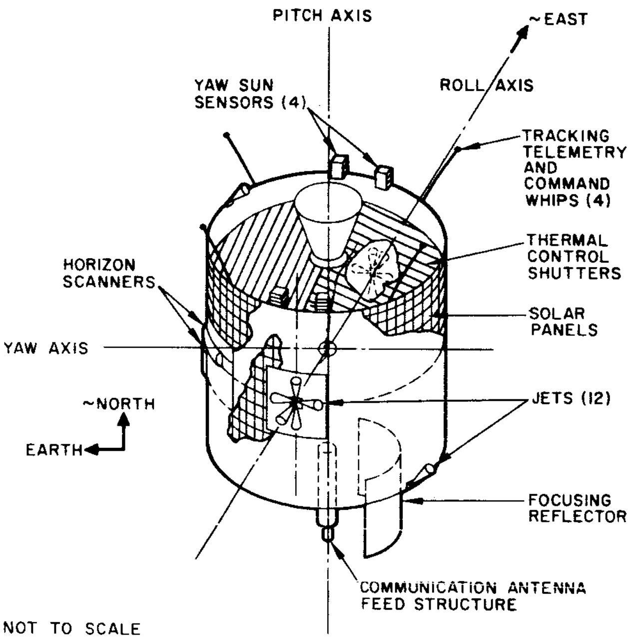


Figure 5-2. Three-Axis Configuration with Body-Mounted Solar Panels

Choice of Sensors

Since the above design concept involves the use of the earth vertical and sun vector as primary attitude references it is natural to select horizon scanners and sun sensors to track the above references. No attempt will be made to describe these sensors at this point except to indicate that they will be similar in complexity to those of OGO or ADVENT. Figure 5-1 shows eight body-mounted yaw sun sensors (employing simple multiple-slit optics) to detect any deviation of the body pitch-roll plane (about the yaw axis) for the ecliptic plane plus two coarse and two fine paddle-axis-mounted sun sensors to detect pointing errors (about the paddle-axis) between the sun line and the array face normal. In addition, two horizon scanner assemblies (each containing two scan heads) and associated circuitry will be used to detect pitch and roll deviations from the local vertical (spherical earth). Additional inhibit signals are needed when the sun enters the horizon scanner field of view. Furthermore, one yaw and one roll rate gyro are needed during de-spin and earth acquisition modes as indicated in Figure 5-1 and discussed below. The yaw rate gyro may also be used to control the body motion about the yaw axis when a noon, midnight, or eclipse condition exists, resulting in loss of a yaw axis reference to the sun. The rate gyros will probably be of the spring restrained, temperature compensated type. The full scale output of the rate gyros will be of the order of 5 deg/sec with an uncertainty of ± 0.05 deg/sec. The yaw gyro will saturate during most of the de-spin mode (starting at about 600 deg/sec) but this is not serious since all system control is operated in a bang-bang mode with suitable deadbands to match the desired low threshold uncertainties of the sensors and hence accommodate rapid convergence from gas jet to reaction-wheel operation.

Polarization measurements of the communication signal mode at a ground station may be used to ascertain and control yaw angle during loss of sun sensor yaw signal (e.g., noon, midnight).

Modes of Operation

Apogee Thrust Vector Control (Mode I). By far the simplest expedient available to control the thrust attitude during apogee motor firing is to use the orient-spinup-Agena-separate concept at perigee of the transfer ellipse, as planned in Syncom II. Otherwise the sensor, acquisition logic, and torque control configuration of the spacecraft will have to be unnecessarily complicated in order to be able to establish and maintain the proper thrust vector attitude for the apogee boost mode (i.e., yaw axis horizontal and inclined to the equatorial plane for removal of transfer orbit inclination) and remove about 50 to 100 ft-lb of misalignment torque during the firing period of about 45 seconds (requiring control jet thrust levels of 12.5 to 25 pounds, assuming a moment arm of about 2 feet and two jets per axis operating at once). Although a minimum of 4.33 pounds of hot gas propellant ($I_{sp} \approx 260$) would be expended as opposed to 1.6 pounds of propellant needed to de-spin the spacecraft (in about 40 seconds) from an initial 100 rpm (with a spin axis moment of inertia of 70 slug-ft^2) using the existing yaw control jets, the above statements are not intended to favor a spin-boost-de-spin approach on the basis of a weight saving. To do so would require a weight comparison of the Agena spin-up system (~ 65 pounds) in terms of its effect on the synchronous orbit payload increment with that of the extra sensor, power, and thrust control components needed to comprise a thrust vector control system. Rather, the complexity associated with adding to or extending the present sensor and logic circuitry is the qualitative criterion (subject to further study) used to select the spin averaging thrust vector control mode. The additional constraint imposed by this mode is that the spacecraft mass distribution be such that the ratio of yaw to roll (or pitch) moment of inertia is $I_z/I_x \geq 1.2$, to bound the subsequent burnout nutation angle to a tolerable value ($\leq$ 1 degree). With the solar paddles in a stowed (folded) position, the mass distribution should be similar to that of Syncom II.

Acquisition (Mode II). The acquisition system and logic are similar to those of OGO (Reference 5-2). The purposes of this mode are to orient the yaw (z_b) axis of the spacecraft close enough to the local vertical and with low enough angular momentum so that horizon scanner control can be obtained, and to orient the normal to the array axis toward the sun. Initial conditions are arbitrary vehicle orientation and rates of up to 1 deg/sec about each axis. Initial acquisition takes place a short time after extension of the solar array paddles and consists of a sun acquisition mode and an earth search mode. Should the horizon scanners later lose the earth, a reacquisition capability is provided, consisting of an array slew mode and an earth search mode with appropriate wheel inhibit signals distributed as shown in Figure 5-1.

Array Slew Mode. The purpose of this mode is to slew the solar array to 90 degrees (normal to array face pointed parallel to body plus roll axis). The body and array mounted sun sensor error signals can then be used to control the yaw and pitch gas jet systems. Following de-spin the array is released at 90 degrees and hence this mode is not necessary in initial acquisition and is bypassed. Following initial acquisition, the array slew mode is entered upon receipt of a loss-of-earth signal from the horizon scanner. (This occurs when two horizon scanners lose lock on the earth.)

Sun Acquisition Mode. The sun acquisition mode is entered at initial acquisition upon receipt of a signal indicating that the solar array paddles are extended. In subsequent acquisitions the sun acquisition mode is entered upon exit from the array slew mode. The purposes of the sun acquisition mode are to align the x-body axis (x_b) toward the sun, reduce yaw and pitch rates to reaction wheel limit cycle rates, and establish a nominal roll rate via the roll rate gyro bias (prelude to earth search mode). In addition, as a result of this roll rotation, the yaw reaction wheel momentum will be limited in value equal to the maximum momentum of the pitch reaction wheel (~ 1 to 1.5 lb-ft-sec). Since the roll axis is fixed in inertial space, the roll rotation causes the yaw and pitch axes to interchange position. Hence if there would be more than the maximum momentum storage in the yaw wheel as a result of this position interchange, the excess will be reduced by the pitch gas jets. This procedure will require a maximum of one revolution of the body about the roll axis. The time it takes to align the roll axis along the sun line and establish the required roll rate (~ 10 minutes) plus the time required for one revolution of the vehicle about the roll axis (~ 25 minutes, roll rate ≈ 0.24 deg/sec) constitutes the total maximum dwell time (~ 35 minutes) in the sun acquisition mode. The exit from the sun acquisition mode to the earth search mode is thus preset to occur 35 minutes after entrance to the sun acquisition mode via a timer signal.

Earth Search Mode. In this mode the control system configuration remains unchanged. The roll axis is kept aligned with the vehicle-sun line and a roll rate of 0.24 deg/sec is maintained. As the vehicle proceeds in orbit the yaw axis is swept through space by the roll rate, and must at some point intersect the earth. Exit from the earth search mode to normal control system operation occurs upon receipt of an earth acquisition signal from the horizon scanner system. Such a signal is obtained when three or more scan heads lock onto the earth and the angle between each scanner head and the $-z_b$ (yaw) axis is greater than a nominal small earth discrimination signal (SEDS ≈ 8 degrees).

The SEDS and the 0.19 deg/sec roll rate requirements are to be obtained as follows: From the geometry of the orbit, sun, and vehicle the minimum time available to see the earth is calculated (~ 17/15 hours). In addition the number of roll revolutions required to assume that the $-z_b$ (yaw) axis (axis of intersection of the horizon scanner planes) intersects the earth at least once during this time is calculated (~ 2-1/8). Combining these results yields the minimum roll rate required to guarantee acquisition in this worst case ($\frac{15}{8} \frac{\text{rev}}{\text{hr}} \approx 0.19$ deg/sec). Adding the roll rate tolerance (± 0.05 deg/sec) to this nominal roll rate gives the maximum roll rate that can occur during earth search (~ 0.24 deg/sec). This maximum rate and the angular acceleration of the roll gas jet system (~ 23 deg/sec^2) in turn give the maximum angular overshoot (< 0.002 degree) that could occur in attempting to remove this rate. This angle sets the minimum earth size that must be discriminated to assure acquisition. Application of the tolerance from the small earth discrimination circuit then gives the nominal and maximum earth discriminated against (~ 8 degrees). Reference 5-3 contains a more detailed description of this mode. The acquisition procedures described above assure that the earth will be acquired in a maximum of one orbital period (24 hours).

Normal Control. The normal control mode is entered upon completion of the earth search mode as indicated by a signal from the horizon scanner logic, which occurs when three or more scan heads have locked onto the earth and the angle between each scan head and the $-z_b$ (yaw) axis is greater than a nominal 8 degrees. Exit from the normal mode to the array slew mode will occur upon receipt of a reacquisition signal if it persists for some time (~ 3 minutes) after initial indication that two or more horizon scan heads are not tracking.

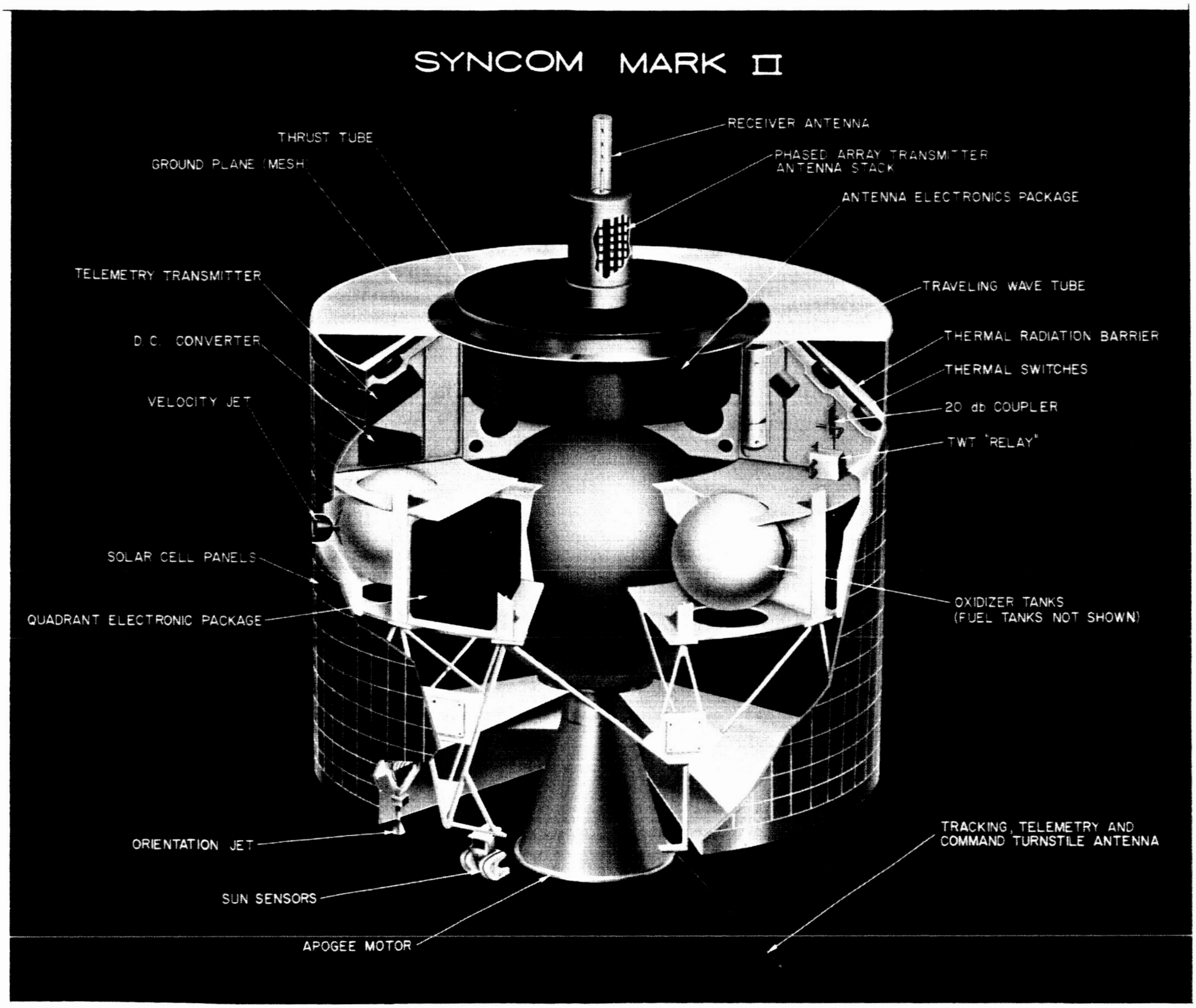


Figure 5-3. Syncom Spin-Stabilized Geometry

The purpose of the normal control mode is to maintain the required vehicle and solar array orientations in the presence of disturbance torques generated naturally or during velocity and orientation commands. As indicated in Figure 5-1, error signals from the horizon scanners control the pitch and roll reaction wheel and gas jet system to keep the z axis of the body diverted toward the center of the earth. In normal control the yaw gas jets are disabled and the yaw reaction wheel system is driven by the error signal from the yaw sun sensors (body mounted). The yaw angle (up to ± 23.6 degrees) required to permit an array rotation to maintain the array normal to the sun is a function of the relative sun angle, which is in turn a function of orbit position and time of year. Once the earth has been acquired, the solar array has two degrees of freedom, one about the vehicle z axis (yaw) and the other about the array axis (pitch). The array face must be oriented perpendicular to the sun to obtain maximum solar efficiency.

An additional constraint upon the array control system may be that the total range of array rotation be no more than 180 or 360 degrees. This limitation makes possible the use of flexible leads rather than slip rings for electrical transmission from array to body. Since the average array rate is equal to earth rate (0.0042 deg/sec), a deadband of ± 0.5 degree would allow as much as 240 seconds of zero motion between slip ring contacts (if they were used) and still limit the duty cycle of the drive motor to (hopefully) a tolerable value. The probability of a cold weld occurring in a hard vacuum during this time is unknown quantitatively for suitable materials, such as steel on graphite, but appears to be low. However, crystalline materials such as graphite have not been sufficiently qualified in a space environment. More conventional materials such as copper-bronze and steel have higher probabilities of cold welding especially during the more than 5 hours of ascent to synchronous altitude. Thus, from a reliability viewpoint it is safer to complicate the control logic (requiring either a paddle unwind sequence at midnight or two noon yaw turns, one at noon and one at midnight, when the array face normal is parallel to the yaw axis) and avoid the use of slip rings especially for a long-life vehicle such as Syncom.

Finally, during the periods of orbital control in longitude (East-West) or inclination (North-South), the spacecraft must be rotated about the yaw axis so that the roll-yaw plane is in the equatorial plane to make the thrust axes tangential to the orbital velocity and normal to the orbital plane respectively. This yaw angle change can be as much as 23.6 degrees (inclination of the ecliptic).

Spin-Stabilized Configuration

Since a detailed description and discussion of the Syncom II spin-stabilized configuration is contained in Reference 5-1, only a brief description with emphasis on the control aspects will be taken from the above reference and repeated here for completeness.

The inertially symmetric properties of an equatorial synchronous orbit plus the stationary geometry of the lines of sight to earth stations motivates one to replicate these qualities in the spacecraft design in order to take advantage of the simplifying symmetry. This is done by trading spacecraft-generated continuous physical control of the geometric axes for the time-based electronic control of both thrust vector and Poynting vector directions via ground commanded corrections. A net simplification will result if the effects of all disturbance torques are rendered small enough (by spinning) to make the attitude correction rate from the ground negligible compared with the orbital stationkeeping correction rate. Since ground commanded stationkeeping corrections are mandatory in both (presently envisioned spin-stabilized and multiaxis) control systems (ground tracking must precede orbit determination and correction command), if the component and system reliability potential of the spinning spacecraft phased array control electronics (PACE) plus sun sensors and jets can be shown to be greater than the corresponding measure of a multiaxis system by virtue of operational simplicity, number and type of components, and ease of incorporating meaningful redundancy, then the effort spent on designing the special PACE circuitry will be worthwhile.

General Description

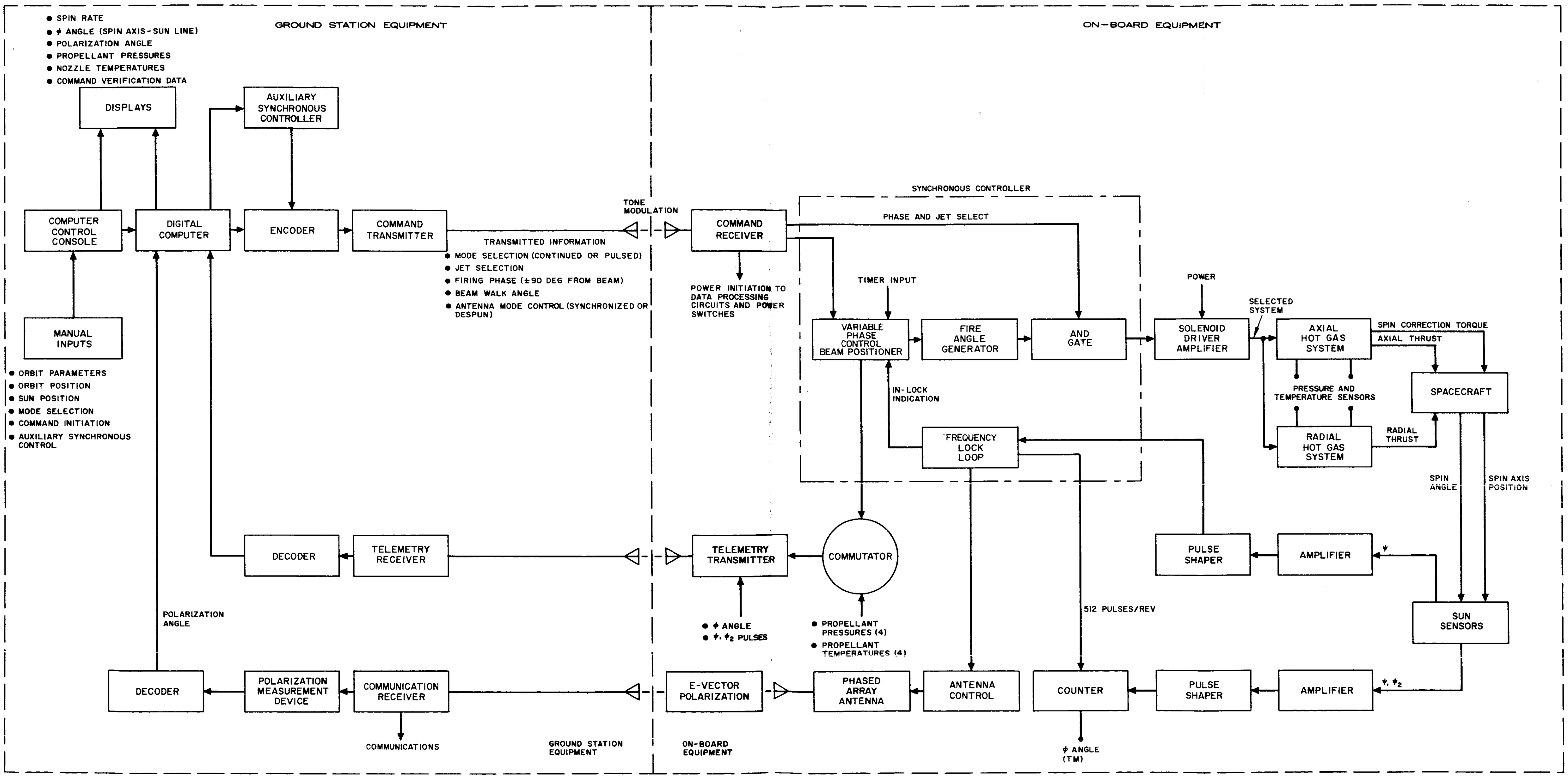
A three-dimensional rendering of the key components of the Syncom II configuration is given in Figure 5-3 with no redundancy shown. The system operation involves both satellite and ground station components as shown in the functional block diagram of Figure 5-4. A discussion of these components as they relate to the functions of the control system is given later.

The Syncom II control system is similar to that of Syncom I. The principal exceptions are:

1) The ground-based synchronous controller for providing properly phased signal pulses to the jets is employed as a backup to a synchronous counter and logic circuitry on board the spacecraft.

2) The two independent reaction jet subsystem units employ bipropellants (MMH and N_2O_4) instead of two separate systems using hydrogen peroxide and cold gas. Two independent and completely identical propellant and engine units are provided, each of which has the capacity to perform all stationkeeping operations throughout the service life.

3) Active spin rate control maintains the spin rate in the range of 100 ± 25 rpm. This is accomplished by means of a centrifugally actuated gimbaled jet with its axis of rotation at 45 degrees to a spacecraft radius. Movement of the thrust vector through a small angle produces a tangential component of thrust of appropriate polarity and magnitude whenever spin rate deviates from the design rpm.



ALL ELECTRONIC CIRCUITS AND SUN SENSORS ARE QUADRUPLY REDUNDANT

PROPELLANT SYSTEMS AND JETS ARE DOUBLY REDUNDANT

Figure 5-4. Syncom II Spin-Stabilized Control System

Syncom II employs four sets of ψ and ψ_2 sun sensors in the same configuration as in Syncom I. These sensors perform the same function as the single set of sensors in Syncom I, that of providing information for determining the spin orientation and for a timing reference to fire the jets in the pulsed mode.

In performing the orientation maneuver, the known initial spin axis orientation at apogee motor burnout is used in computing the phase delay and the total precession angle required to align the spin axis to the earth's polar axis. Errors in the initial orientation are, in general, small, and the resulting error at the completion of the maneuver will be correspondingly small. Measurement of the polarization angle of the energy received from the linearly polarized transmissions of the phased array antenna provides the information for making final correction in spin axis orientation. In the event that large orientation errors exist so that the antenna beam is not detected at the completion of the maneuver, the antenna electronic control circuits may be deactivated, causing the conical beam to revert to a pattern similar to the Syncom I antenna pattern. The included angle of the beam will then intersect the earth and will be detectable as an RF signal. Measurement of the polarization angle together with the sun sensor information will provide sufficient data to determine the orientation of the spin axis.

The physical arrangement of the reaction control jets is similar to that of Syncom I. The redundant radial jets are located on opposite sides of the spacecraft with thrust vectors pointing through the center of gravity. The axial jets are also 180 degrees apart with thrust axes parallel to the spin axis and at a radius of about 26 inches from the spacecraft spin axis.

The control jets are used in either a continuous or a synchronous pulsed mode, depending on the operation to be performed. Table 5-1 presents a list of the operation and the manner in which the jets are used. Also included in the table is the maximum total impulse required in each operation, expressed in units of equivalent velocity increment imparted to the payload.

Functional Description

The block diagram of Figure 5-4 identifies the functional elements of the control system and their interrelationships. The basic functions performed by this system are:

1) To produce a thrust vector in the appropriate direction in space for the required velocity correction

2) To produce a moment about the appropriate axis in space to precess the spin axis in the required direction

A secondary function is to maintain the spin rate within a prescribed range.

TABLE 5-1. MAXIMUM TOTAL IMPULSE REQUIRED

Function	Jet Used	Mode	Equivalent ΔV
Spin axis orientation	Axial	Pulsed	18 fps
Orbit period and eccentricity correction	Radial	Pulsed	120 fps
Orbit inclination correction	Axial	Continuous	112 fps
Stationkeeping, East-West errors	Radial	Pulsed	7 fps/yr
Stationkeeping, North-South errors	Axial	Continuous	180 fps/yr
Solar pressure precession 0.75 deg/yr	Axial	Pulsed	0.11 fps/yr

Velocity correction may be performed with either the axial jet in a continuous mode or the radial jet in a pulsed mode. Selection of the mode depends on the particular type of orbit correction required. The spin axis, in general, will not be reoriented once the initial alignment has been established; however, when required, precession of the spin axis is accomplished by use of the axial jet in a synchronously pulsed mode.

The direction of either the velocity maneuver (when using the radial jet) or the precession maneuver depends on intelligence computed on the ground and transmitted in digital code to a register in the satellite. The magnitude of the maneuver is controlled by the duration of the execute signal from the ground station. Thus with the exception of spin rate control, which is performed entirely by on-board sensing and control, the ground station is an integral part of the velocity and orientation control system.

The basic information required for synchronous pulse jet control is the spin axis orientation and a timing signal to indicate the relative position of the jets with respect to a space coordinate system. The latter signal is provided by the ψ sensor, whereas the orientation is established by the combined ψ and ψ_2 sensor signals, both of which are transmitted in real time to the ground station via the telemetry link.

Orbit corrections based on satellite tracking data are determined at the ground station by a digital orbital correction command computer.

An auxiliary synchronous controller at the ground station, shown in Figure 5-4, provides the capability of controlling the jets in the synchronous pulse mode in a manner similar functionally to the synchronous controller used in Syncom I. However, it is an all-electronic device composed of circuits essentially identical to those used in the on-board synchronous control. Jet on-off commands are transmitted directly from the ground via the execute signal. The on-board register must be set for continuous mode when the auxiliary controller is used.

Sun Sensors

General. Four sets of four sensors will be used per spacecraft, with two ψ and two ψ_2 sensors per set. Signals from one pair of ψ and ψ_2 sensors will be telemetered and those from the second set will be used by the on-board electronic systems. Switching will be provided to allow use of any set.

Signal Strength. With a sun incident angle of 90 degrees, the minimum peak sensor output voltage shall be 250 millivolts. For incident angles of 15 and 165 degrees, the minimum peak sensor output voltage will be 185 millivolts.

Bandwidth. For a sun incident angle of 90 degrees, the angular distance between the 3 db power points, obtained when the sensor is rotated about an axis parallel to its sensing plane, will be 0.80 ±0.10 degrees.

Positive Slope Reference. The sun sensor outputs will be shaped prior to use by the electronic system. The sensor level, triggering the shaping circuit, will be 100 millivolts ±10 percent. The deviation in the angle at which a sensor has an output of 100 millivolts will be within ±0.2 degree of a design angle that will be specified by the vendor.

Alignment of ψ and ψ_2 Sensors. The ψ and ψ_2 sun sensors will be aligned so that the angle between their sensing planes will be 35.0 ±0.5 degrees.

Reference Sensor. In the assembly of four sun sensors, the outer ψ and inner ψ_2 sensors will be designated as references for alignment to the spacecraft. These will also be utilized by the on-board electronics.

Synchronous Controller

General. The function of the on-board synchronous controller is to control the firing of the reaction control jets. It forms a part of the electronic system associated with the phased-array antenna. The portions of the phased-array electronics used for this function are the low-frequency multiplier, variable phase control, and fire angle generator. The low-frequency multiplier utilizes the ψ sensor output to provide 512 counts per spacecraft revolution between ψ pulses. Ground command inputs are inserted into the variable-phase control circuits to provide a jet firing position relative to the sunline. The fire angle generator receives inputs from the frequency-locked loop and variable phase control, as well as initiating firing and jet selection commands from the ground to activate a power switch that operates the jet control valves.

Commands. The following ground commands will be required by the synchronous controller.

1) Communication beam walk angle for firing the jets at other than 90 or 270 degrees from the beam

2) Firing phase - 90 or 270 degrees

3) Mode select - continuous or pulsed

4) Jet selection - four jets

5) Antenna mode control - either synchronized or omnidirectional. This allows antenna information to be used to aid in the orientation maneuver if needed.

Sun Sensor Amplifier and Pulse Shaper. With a ramp input into the sensor amplifier of 10 v/sec, the shaping circuit will operate between 90 and 110 millivolts. The time lag between sensing of the proper activation signal and maximum output voltage from the shaping circuit will be less than 100 microseconds.

Angular Resolution. All angular references and commands utilized by the fire-angle generator in determining the jet firing angle will have a resolution of at least 0.70 degree.

In-sync Interlock. An in-sync condition, defined by the frequency lock loop operating at 512 ±1 counts per revolution, will be required to exist coincidently with the initiated command to actuate the jet control valve.

Operating Time. The jet pulse controller must be capable of operating continuously once for at least 1.5 hours.

Alignment of Components on Spacecraft

Reaction Control Jets. Two axial and two radial jets are required per spacecraft. The axial jets will be placed diametrically opposite one another, as will the radial jets. The radial and axial grouping of jets will be 90 degrees apart.

One radial jet will be designated as a reference. Alignment of the other jets will be 22.5 and 202.5 degrees ±0.25 degrees from the reference jet. The alignment point on each jet will be the geometrical center of the jet nozzle.

The geometrical centerline through the jet nozzle will be perpendicular to the spin axis of the spacecraft and intersect it at a position defined as the cg position for the condition of a burned-out apogee motor, within an angle of ±0.25 degree.

The axial jets provide spin-speed control, as well as precession torques. The jet rotates under the influence of centrifugal force about an axis nominally 45 degrees to a spacecraft radius. Scribed lines on the base of the jet, indicating the center of the rotational axis, should be aligned 45 ±0.5 degrees to a spacecraft radius. The base should be within 0.50 degree of being perpendicular to the spacecraft spin axis.

Sun Sensors. Four sets of four sun sensor assemblies are mounted around the periphery of the spacecraft. They will nominally be 90 degrees apart.

The sun sensor assemblies will be placed around the circumference of the spacecraft at 45, 135, 225, and 315 degrees ±0.25 degree, relative to the reference radial jet. The outer ψ sensor in the assembly of four sensors will be used for aligning.

The sensing plane and leading edge of the reference ψ sun sensor will be parallel to the spin axis within ±0.50 degree. The sensing plane will lie within ±0.25 degree of a radial line of the spacecraft.

Electronics

Figure 5-5 is a block diagram of the phased array control electronics (PACE) and jet control electronics subsystem. This diagram incorporates the frequency lock loop (FLL), waveform generator, and the ψ_2 counter that counts and stores the number of cycles of the frequency lock loop-voltage controlled oscillator between the ψ and ψ_2 pulses. The contents of the counter are telemetered as digital information, providing the ψ - ψ_2 angle to an accuracy of ±0.35 degree. The variable phase control subassembly is now called the beam positioner subassembly.

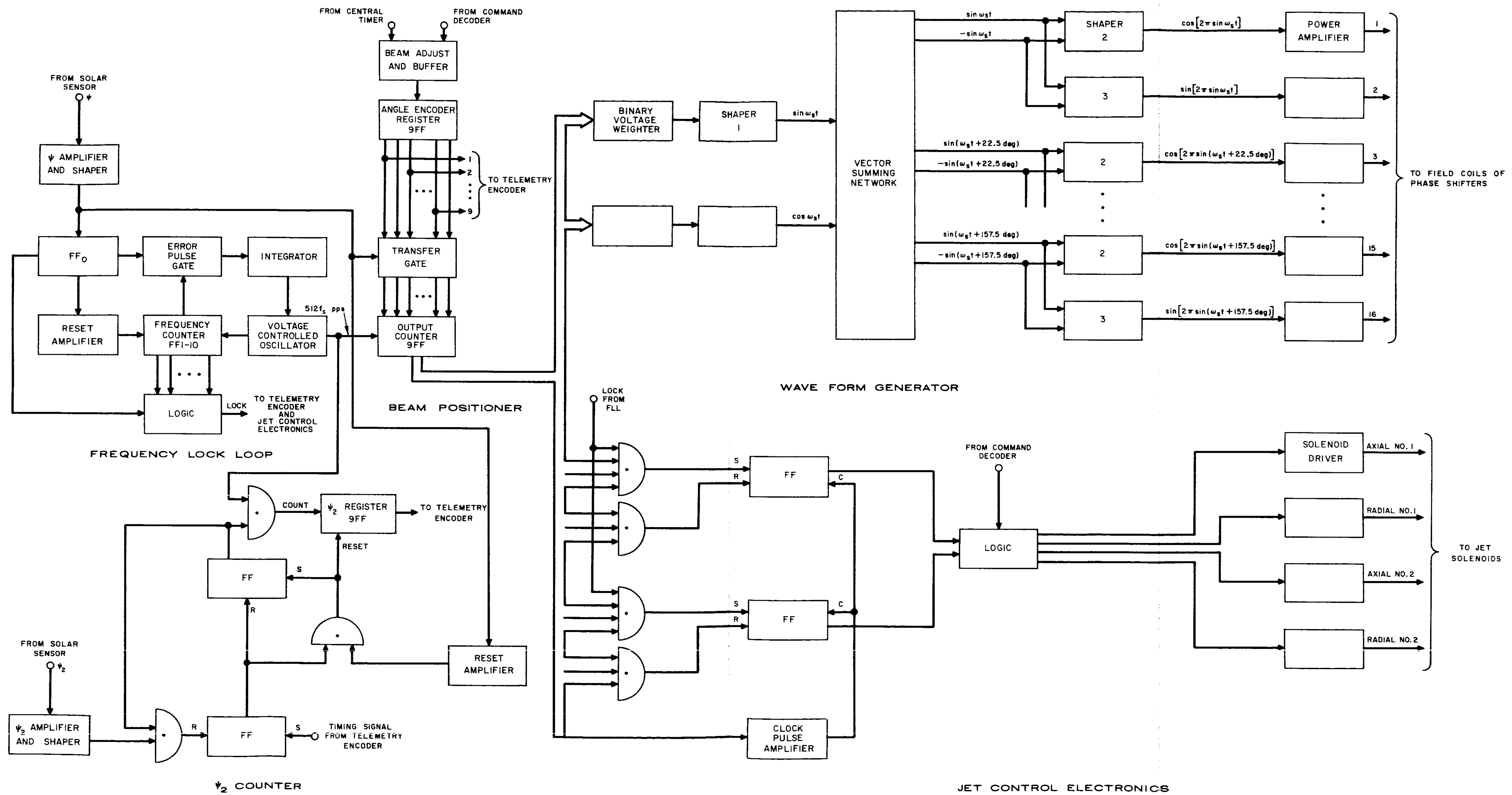


Figure 5-5. Phased-Array and Pulse Jet Control Electronics

Frequency Lock Loop (FLL). The FLL generates the reference timing signals for the PACE. Its output is a square wave, the frequency of which is 512 times the spin frequency, f_s. An auxiliary output is a digital signal that indicates when the FLL is in lock, that is, when count = $(512 \pm 1) f_s$.

Jet Control. The advanced jet control subsystem will be designed to provide a 45-degree pulse envelope such that the average thrust direction will be displaced ±90 degrees from the earth-spacecraft line by sequential timing and a continuous pulse envelope in real time. The sequential timing portion of the advanced subsystem will enable pulsing of both radial or both axial jets in such a manner that the pulse envelope per revolution of one jet is displaced 180 degrees relative to the pulse envelope per revolution of the other, or that only one jet is activated with its pulse envelope occurring once every spacecraft revolution.

The block diagram of the advanced jet control subsystem is shown in Figure 5-6, and a simplified spacecraft configuration illustrating the jet positions is shown in Figure 5-7 .

The characteristics of the variable phase control output counter are such that the counter has a count of zero when the reference (zero phase shift) ferrite phase shifter element is coincident with the spacecraft-earth line.

The backup mode provides the capability of pulsing the jets in real time if the sequential timing circuits fail. Also, the backup mode can be used to operate a jet continuously.

The solenoid coil amplifiers will be designed with series output transistors and each driven by a separate preamplifier. Each preamplifier can be controlled by the input signal. The series redundant configuration decreases the probability of a solenoid coil's being continuously activated without a command, since to close the coil circuit requires that both onput transistors fail.

Waveform Generators. Waveform generators for the phase shifters are implemented with the use of a combination of greater and lesser gates. Zener diodes are used to shift the signal's dc bias as necessary as the signal passes through the circuit.

Comparative Discussion

Control Components

From the above brief description of the two system configurations and from Figures 5-1, 5-4, and 5-5 it can be seen that a conventional multiaxis control system requires from 10 to 12 gas jet assemblies and

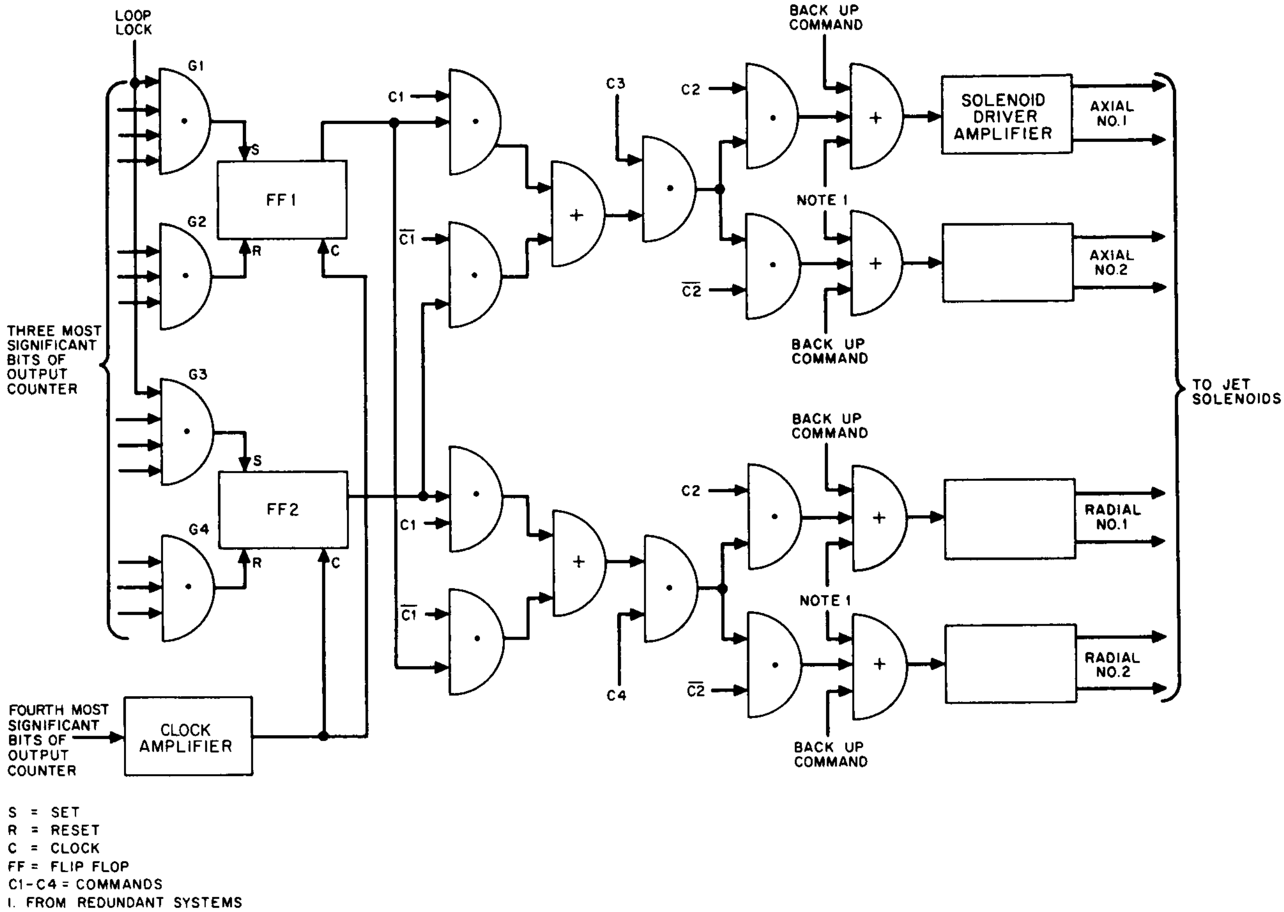
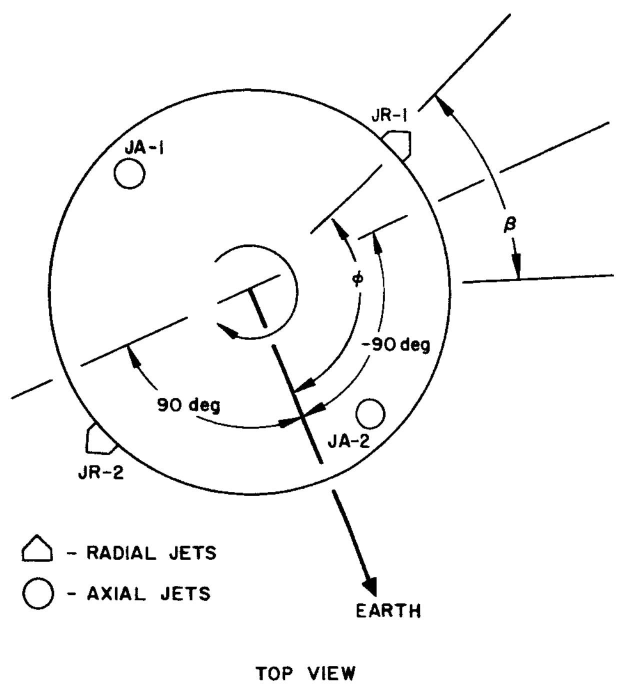


Figure 5-6. Jet Control Electronics



β = 45 deg ANGLE THROUGH WHICH JET IS ACTIVATED

ϕ = FIXED-ANGLE BETWEEN REFERENCE FERRITE PHASE SHIFTER AND JR-1

JR = RADIAL JET

JA = AXIAL JET

Figure 5-7. Spacecraft Configuration

three reaction wheel assemblies to accommodate the required attitude and velocity control as opposed to only two jet assemblies for the spin-stabilized Syncom. Some preliminary investigations at Hughes indicate that with the use of some ingenuity a six-jet, three-wheel configuration can be shown to be adequate for three-axis attitude and velocity control. But it is more than just the disparity in the number of jets that favors the spin-stabilized approach in the control component comparison; rather, it is the required use of rotating components (reaction wheel motor, tachometer) in the multiaxis system and (more important) the inherent advantage of control component redundancy in the spin-stabilized system that favor the latter. Although the reaction wheel motor can be designed to operate at reasonably low speeds for this application (approximately 1000 rpm maximum) so that an extrapolated ball bearing life of 3 to 5 years may be available, the development and qualification of such a device has not yet been accomplished for such a long period. For bearing life greater than 5 years other techniques such as the use of mercury flywheels or externally pressurized air bearings appear more promising but are not now available. In other words, the proper qualification of rotating components for long-life space operations appears to be lagging behind that of solid-state circuit elements. This is partly due to the nature of the qualification (long testing times) and partly to the relative lack of impetus within the industry in this direction until the more recent advent of space vehicle attitude control requirements using reaction wheels.

In attempting to use redundancy at the component and subsystem level, a dual set of jets and fuel tanks (with appropriate cross-feed) would tend to complicate the switching logic and result in a non-negligible weight penalty in a multiaxis system because of the number of jet assemblies. Moreover, should one jet fail open, an additional equal amount of fuel will have to be expended to counteract the subsequent motion, resulting in no real gain from redundancy until one of the redundant sets of tanks feeding the failed jet is depleted or shut off. In the spin-stabilized system, in addition to the relative ease of incorporating two separate control units, should one of the radial jets fail open, the motion of the spacecraft would average to zero in one spin period (approximately 0.6 second). Should one of the axial jets fail open, the motion of the spacecraft would average to zero in one orbital period (24 hours) if nothing is done. Such a failure would not upset the attitude (and hence the communication link) of a spinning configuration if not immediately corrected.

Sensors

Sun Sensors. Although the sun sensor elements are comparable in both configurations the functional use of these elements in the spin-stabilized system is simply to generate a time reference for spin phase and spin axis attitude determination, whereas actual angle measurements must be made in two planes in the multiaxis system and compared with a yaw body axis and array normal reference direction to generate appropriate error signals for the control system, as indicated in Figure 5-1. This requires a greater number (because of shadowing) and more complex sun sensor assemblies and processing and logic circuitry (similar to OGO) than in the spin-stabilized design, in which simplified slit optics and an amplifier-shaper are adequate.

Gyros. Because of the inherent uncertainty of the attitude of a non-spinning vehicle upon emergence from an eclipse condition or possibly after de-spin, a roll rate gyro is needed to generate the proper roll rate during the earth search mode as described above. A yaw rate gyro is needed to monitor the yaw motion during the de-spin phase (and control yaw motion during noon turns or eclipse conditions). Although body-mounted yaw sun sensors may be used in a rate sampling mode during the onset of the de-spin phase, the sample rate will become too low for adequate control (and signal differentiation too noisy) as the spin speed approaches zero. The duty cycle of these gyros can be made low by switching them on only when needed; nevertheless, they still require accompanying spin motor power, a pickoff amplifier, and demodulator electronics. No gyro assemblies are needed in the spin-stabilized system.

Horizon Scanners. Since a multiaxis system must be constantly controlled to track the local vertical, the horizon scanner is a key sensor for this configuration. Although a number of suitable designs exist, they must each embody a search and track function, which implies that a mechanical scanning mechanism must be adapted to operate continuously and for a long time in a space environment. The scanner assemblies, electronics, and associated signal processing and mode logic circuitry may not be uniquely complex but must accomplish a number of functions. A brief description of the OGO horizon scanner (Reference 5-2) is given at the end of this section (section 5) to indicate what it takes to make one work and to balance some of its electronics against part of the PACE circuitry in the spin-stabilized Syncom.

In addition, provisions must be made to yaw the spacecraft when the sun appears on a horizon scanner. When the sun enters the field of view of a scanner, a sun interference signal is available from the horizon scanner logic (Figure 5-1). If such a signal is received from scanner C and there is no negative (CCW) drive voltage or no drive voltage applied to the yaw reaction wheel motor, then a positive (CW) drive voltage is applied to yaw the scanner away from the sun. If such a condition exists and another scanner has failed, the pitch and roll control systems are inhibited until scanner C is yawed away from the sun. Should the sun appear in the field of view of scanners B or D, normal operation of the yaw control system will drive them off of the sun. If the sun appears in the field of view of scanner A, the logic to eliminate an unstable null will cause the scanner to be driven off of the sun. In order to avoid a reacquisition during this time, the pitch and roll control systems are inhibited. Further, a reacquisition signal must persist for at least 2.9 minutes before a reacquisition is commanded. This is sufficient time to make all but a negligible percentage of turns.

No horizon scanners are needed in the spin-stabilized system and no special logic is needed to cope with special sun positions (except for a reasonable launch window restriction).

Electronics

In order to make the comparative discussion of attitude control components complete one must include the PACE circuitry (described above) used to de-spin the communication pencil beam in the spin-stabilized system. Since this function is unique to the spin-stabilized configuration no functional comparison will be practical except to recall that the nonmechanical, solid state, low power signal level components of the PACE assembly provide the equivalent pointing function of the mechanical assemblies (reaction wheel motors, gyros, horizon scanners) and associated spaceborne electronics needed in a multiaxis, closed-loop attitude control system. To show that the mechanical motion of the multiaxis attitude control components and their associated electronics consume more or less power than their spin-stabilized functional counterparts would require a rather detailed design of a multiaxis system suitable for a Syncom mission. Even then the task of apportioning the proper fraction of the total system power to the attitude control function would be difficult. Rather, an attempt will be made to determine (roughly) the total system power requirements for a multiaxis communication satellite by scaling from some STL recommended solar array areas for the ADVENT satellite. These results will then be compared with the present Syncom II power requirements. Although a smaller total power consumption is not completely synonymous with increased system reliability potential, it is definitely indicative of superior component reliability for a system of comparable complexity.

System Power, Solar Array Efficiency

The recommended single-paddle area for the ADVENT system using 2-watt traveling-wave tubes (TWT) is about 15.1 square feet, or a total array area of A_2 = 30.2 square feet. Now, from Syncom II design values the ratio of TWT power requirement to total system power (using 4-watt TWTs) is about (73/124 = 0.59). Assuming (optimistically) a similar ratio ($\geq$ 0.2) for a multiaxis 2-watt TWT system design, since solar array area is proportional to available power, the multiaxis system array area A_4 may be estimated for 4-watt TWTs as

$$A_4 \geq (1.2)(A_2) = (1.2)(30.2) \cong 35.6 \text{ square feet}$$

Using a solar constant of 130 watt/ft^2, N-P cell efficiency of 9 percent, the electric power available P_a with no loss is

$$P_a \approx (0.09)(130)(35.6) \cong 416 \text{ watts}$$

Allowing for 8 percent degradation (radiation aging) the effective power P_e available is

$$P_e \approx (1-0.08)(416) = 382 \text{ watts}$$

Finally, allowing a 12-watt safety margin (as in Syncom II) gives the estimated system required power, $Pr_o \cong 370$ watts for a multiaxis system, compared with $Pr_s = 124$ watts for Syncom II ($\sim 1/3\ Pr_o$). Although the above estimate for Pr_o is crude, it is believed to be optimistic, since data taken from References 5-1 and 5-4 and presented in Table 5-2 shows a total power requirement of 548 watts for ADVENT. The subsystem power requirements for the multiaxis estimate is scaled approximately as ADVENT so that even with 218 watts allowed for communications the attitude control allotment of 80 watts is significantly greater than the 17 watts of the PACE circuitry.

TABLE 5-2. SYSTEM AVERAGE POWER REQUIREMENT ESTIMATE

Subsystem	ADVENT	Multiaxis Estimate, watts	Syncom II
Communications (four channels)	300	~218	80.9
Track, telemetry, and command	34	~ 23	10
Attitude control	132	~ 80	(PACE) 17.1
Instrumentation	12	----	----
Operating total	478	~321	108
Battery charging	70	~ 49	16.3
Total power required	548	~370	124.3

One of the disadvantages of body-mounted solar cells on the cylindrical surface of Syncom II is the resultant geometric reduction in efficiency by a factor of $1/\pi$. The present Syncom II solar cell array plus supports weighs 43 pounds and yields 135 watts under the worst sun incidence angle (25 degrees) and after adjusting for 8 percent degradation. Using the solar paddle density of 0.0345 slug/ft^2 from Surveyor studies, the solar array paddles for the multiaxis configuration should weigh (35.6)(0.0345) ≅ 1.23 slugs = 39.8 pounds. Thus the array weight penalty incurred by the poor illumination geometry on the Syncom II design is almost compensated by the more efficient attitude control concept requiring much less total power. This compensation is even more favorable toward Syncom II if one included the weight of the solar array paddle drive motor and gear train assembly (~3 to 5 pounds) (shown schematically in Figure 5-1) as part of the multiaxis array weight. Furthermore, for a given solar panel area failure the subsequent Syncom II average power reduction is only $1/\pi$ times the power reduction of a multiaxis planar array.

It is indeed not surprising that a spin-stabilized system consumes less power when one considers the fact that attitude angle error accrues as the product of torque and time of application, whereas this same product will generate an angular rate which must be continuously bounded in a multiaxis system during the entire lifetime even if the torque level is small. More precisely, for a constant torque $\overline{N}$ applied to a spinning vehicle with angular momentum $\approx I_S\overline{\omega}_S$ (I_S = moment of inertia about spin axis, ω_S = spin angular velocity), the resultant body precession rate $\overline{\omega}_p$ obeys the expression (for $\overline{N}$ normal to $\overline{\omega}_s$).

$$\overline{N} = \overline{\omega}_p \times I_s\overline{\omega}_s = \omega_p I_s \omega_s \tag{5-4}$$

If the same torque is applied to a nonspinning body the motion parallel to $\overline{N}$ will obey the equation

$$\overline{N} = I_N \dot{\overline{\omega}}_o \tag{5-5}$$

where $\dot{\overline{\omega}}_o$ = the angular acceleration of the body about an axis parallel to $\overline{N}$. Now, if $I_N \approx I_S$ (a reasonable assumption for either configuration), expressions 5-4 and 5-5 may be equated; i.e.,

$$\dot{\omega}_o \approx \omega_p \omega_s \tag{5-6}$$

Setting $\omega_o(o) = \omega_p(o) = 0$ and integrating once gives

$$\omega_o = \omega_p(\omega_s t) = \omega_p \theta_s; \quad t > 0 \tag{5-7}$$

where θ_S is the spin angle accrued in time t. Integrating a second time shows that during an interval t the nonspinning body will have rotated an amount θ_O given by

$$\theta_o \cong \theta_p(\omega_s t) \tag{5-8}$$

where $\theta_p = \omega_p t$. Since the largest disturbance torque appears to be solar radiation pressure unbalance, which precesses Syncom II at the maximum rate of 0.75 deg/yr, the equivalent uncompensated nonspinning body rate will be of the order (from Equation 5-7)

$$\omega_o \approx (0.75 \text{ deg/yr})(10.5 \text{ rad/sec})(3.15 \times 10^7 \text{ sec})$$

$$\approx 2.5 \times 10^8 \text{ deg/yr} \approx 7.85 \text{ deg/sec}$$

at the end of 1 year. The angle θ_o accrued by the nonspinning body is thus quite large if not continuously corrected. The attitude correction rate for Syncom II, on the other hand, is almost negligible. This fact is in itself an adequate justification for commanding this correction from the (otherwise mandatory) ground control station with a net simplification in the spacecraft attitude control design.

Conclusions

1) Either the spin-stabilized or the multiaxis configuration can be designed to initially meet the performance requirements of the Syncom mission with present state-of-the-art components.

2) The required use of ground control stations for Syncom orbital control also favors the use of this ground control link for attitude correction only if the attitude correction rate is small and the resultant spacecraft simplifications are significant.

3) The total spacecraft power requirements of a spin-stabilized system are significantly less than those of a multiaxis configuration but Syncom II requires a solar array weight comparable to a multiaxis design.

4) The proved reliability potential of most of the critical sensor and control components in the spin-stabilized design are equal to or better than that of the multiaxis design for the 5-year Syncom mission.

5) The nature of the spin-stabilized system allows a more facile and meaningful use of redundancy in the control subsystem when compared with the required use of parallel control elements in the multiaxis design with no redundancy.

6) The spin-stabilized mode control logic appears to be simpler than the sensor and mode control logic of a multiaxis system, especially during periods of eclipse, initial acquisition, and orbital correction.

EARTH SENSOR FOR SYNCOM

In a spin-stabilized spacecraft, the angle between the spin axis and sunline may be readily obtained by use of properly oriented slit-type sun sensors. One additional reference is required to ascertain the pointing direction of the spin axis; it may be found by measuring the polarization angle of transmitted linearly polarized electromagnetic waves from the spacecraft. Such waves, however, are subject to Faraday rotation upon entering the earth's upper atmosphere and the magnitude of this effect is not constant. Another method is using optical sensors to scan the earth or some other celestial body for the second reference. Since the earth subtends the largest angle to a synchronous vehicle it is the logical choice. The feasibility of using an on-board sensor to determine the angle between the spin axis and the spacecraft-earth line is examined in this study. The use of appropriate automatic inhibit-logic circuitry to preclude erroneous interference signals from the sun and moon is not considered.

Wavelength Considerations

The amount of light in the visible spectrum reflected from the earth will vary considerably with angle of illumination by the sun and cloud cover (Reference 5-5) and is therefore not suitable for the purpose. The infrared portion of the spectrum suggests itself as a more uniform source of radiation. If the detector responds to a wide range of the infrared spectrum (i.e., from 1 to 20 microns), the earth appears as a blackbody energy source, varying from about 210 to 300°K, depending upon cloud cover. If the detector output is proportional to incident energy, then according to the Stefan-Boltzman law the ratio of output between the cold and hot portions is

$$\frac{Ec}{Eh} = \left(\frac{210}{300}\right)^4 = 0.24$$

This nonuniformity in source temperature results in errors in the determination of the horizon.

If filtering is used to limit wavelengths from about 14 to 18 microns (Reference 5-9), the earth appears as a substantially uniform source. A system utilizing this bandpass will therefore be proposed.

Principle of Operation

The proposed system consists of two body-fixed sensors, mounted with their optical axes coplanar with the spin axis and looking approximately radially as shown in Figure 5-8. Each sensor has a field of view approximately 1 by 1 degree and the optical axis makes an angle $\theta/2$ with the satellite equator. The angle, θ, is chosen less than 17 degrees so that when the spin axis is nearly normal to the earth line, both sensors will see the earth each satellite revolution. θ is made large enough so that neither the sun nor the moon will simultaneously appear in both fields of view. For reasons discussed later, a value of ~13 degrees has been chosen. When the satellite spin axis is normal to the earth center-satellite line, both fields of view intercept the earth's horizon simultaneously. Any difference in the time of intercept is a measure of departure from the desired condition, and the direction of the error is obtained from a knowledge of which sensor crosses the horizon first. This information then can be used on board (for antenna pointing for example) or telemetered to earth.

For the purposes of this feasibility study, the characteristics of an existing sensor were used. Barnes Engineering Company (References 5-6 through 5-9) has developed a horizon sensor for meteorological satellites which utilizes a 3/4-inch aperture and a germanium immersed thermistor detector. The active flake is only 0.1 millimeter square, providing a field of view of about 1.3 degrees. The unit contains a transistor amplifier, is about 1 1/2 by 1 1/2 by 5 inches and weighs 8 ounces. It does not necessarily represent an optimized design.

Sensitivity

Assume that in the spectral band $\Delta\lambda$ the earth and/or atmosphere effectively radiate at blackbody temperature $T^\circ K$. If

$$N(T^\circ) = \text{total blackbody radiance } (w/cm^2\,\Omega) \tag{5-9}$$

and

$$\eta = \text{spectral utilization factor (fraction of total energy in } \Delta\lambda)$$

then the "effective" irradiance of the aperture is

$$H' = N\eta\Omega \ (w/cm^2) \tag{5-10}$$

where

$$\Omega = \text{solid angle of field of view (steradians)}$$

The effective flux through the optics is

$$F' = H' A_o \rho \ (\text{watts}) \tag{5-11}$$

where

$$A_o = \text{area of objective } (cm^2)$$

$$\rho = \text{transmissivity of optics}$$

The noise - equivalent-power (flux on detector to produce $S/N = 1$) of the detector is related to detectivity as (see Figure 5-9).

$$NEP = \frac{(A_d \Delta_f)^{1/2}}{D^*} \text{ (watts)} \qquad (5\text{-}12)$$

The maximum D^* for a thermistor-bolometer in the usual bridge circuit (active + compensating flake) is

$$D^* = 0.8 \times 10^8 \sqrt{\tau} \text{ (cm cps}^{1/2}\text{/w)} \qquad (5\text{-}13)$$

where τ = time constant in milliseconds

The S/N (rms) is therefore

$$S/N = \frac{F'}{NEP} \qquad (5\text{-}14)$$

Substituting from Equations 5-10 through 5-13

$$S/N = \frac{N \eta \Omega A_o \rho D^*}{(A_d \Delta f)^{1/2}} \qquad (5\text{-}15)$$

Assuming

$$\Delta\lambda = \lambda_2 - \lambda_1 = 18 \text{ to } 14 \text{ microns } (CO_2 \text{ band})$$

$$T = 210°K \text{ (effective blackbody temperature)}$$

then

$$N = 3 \times 10^{-3} \text{ watts/cm}^2 \Omega$$

$$\eta = 0.18 \text{ (210°K, 14 to 18 microns)}$$

If the sensor has an aperture of 0.75 inch diameter (d_o),

then

$$\Omega = \alpha^2 = \left(1.3 \text{ degrees} \times 1.75 \times 10^{-2}\right)^2 = 5.1 \times 10^{-4} \text{ steradians}$$

If the field of view = 1.3 degrees = α (square)

$$(\text{Area detect})^{1/2} = 0.1 \text{ mm} = (A_d)^{1/2} = 10^{-2} \text{ cm}$$

$$A_o = \frac{\pi}{4} d_o^{\,2} = \frac{\pi}{4} [0.75 \times 2.54]^2 = \frac{\pi}{4} (3.6)$$

$$A_o = 2.8 \text{ cm}^2$$

If the detector $\tau = 2.5 \times 10^{-3}$ seconds

$$D^* = 1.26 \times 10^8 \text{ cm cps}^{1/2}/\text{w}$$

$$\rho = 0.3 \text{ (includes bandpass filter)}$$

$$\Delta f = 100 \text{ cps}$$

Substituting into Equation 5-15

$$S/N = \frac{(3 \times 10^{-3})(0.18)(5.1 \times 10^{-4})(2.8)(0.3)(1.26 \times 10)}{(10^{-2})(10)}$$

$$S/N = \frac{2.9 \times 10}{10^{-1}} = 290 \tag{5-16}$$

<u>Analysis</u>

If the spin axis is normal to the earth vehicle line, which is the desired orientation, both fields of view will simultaneously intersect a line of the earth. For small angular deviations from the desired orientation, there will be a time difference of this intersection. The geometry of the situation may be illustrated as follows. Consider for the moment the fields of view of the sensors are of negligible angular dimension. From a consideration of Figure 5-10.

$$d = r(1 - \cos \beta/2) \tag{5-17}$$

$$\cos \beta/2 = \sqrt{1 - \left(\frac{a}{r}\right)^2} \tag{5-18}$$

$$d = r(1 - \sqrt{1 - \left(\frac{a}{r}\right)^2} \tag{5-19}$$

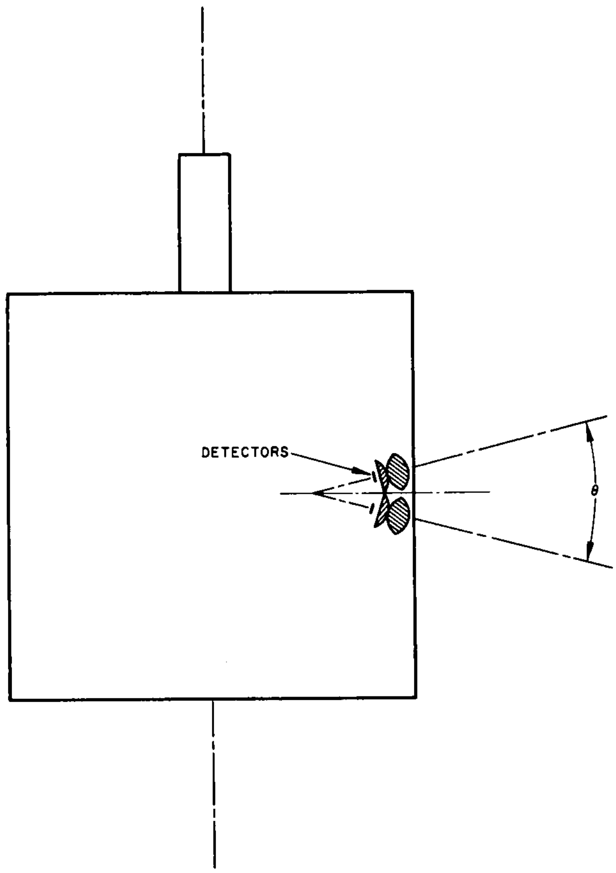


Figure 5-8. Earth Sensor Mounting Geometry

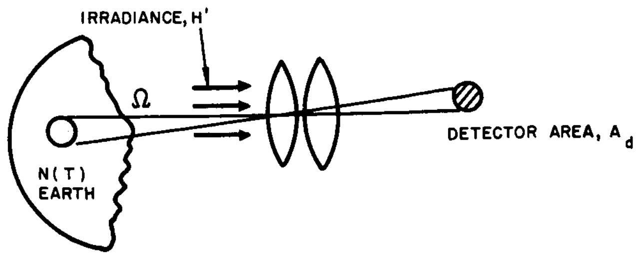


Figure 5-9. Optical Geometry

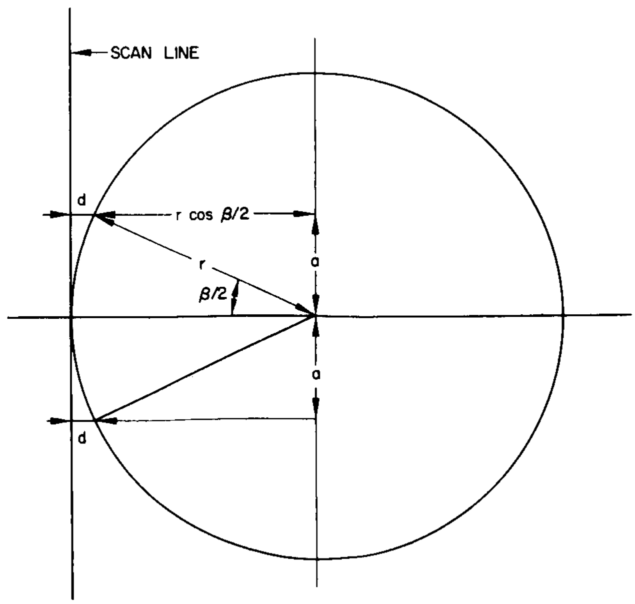


Figure 5-10. Geometry for Analysis

$$\frac{\Delta a}{2\Delta d} = \frac{r}{2a}\sqrt{1-\left(\frac{a}{r}\right)^2} \qquad (5\text{-}20)$$

which represents the relationship between the time difference of pulses and the pointing error.

Consider a square aperture of the above dimensions scanning across a uniform earth. The relative sensor output is plotted as a function of angle for various chords (Figure 5-11) in the absence of time constants, in the detector or amplifier. These functions may be approximated by ramp functions.

$$G_{DA} = \frac{1}{\tau_1 \tau_3}\left[\frac{s}{\left(s+\frac{1}{\tau_1}\right)\left(s+\frac{1}{\tau_2}\right)\left(s+\frac{1}{\tau_3}\right)}\right] \qquad (5\text{-}21)$$

where τ_1 is the cell time constant and τ_2 and τ_3 the lower and upper amplifier time constants. The values for these parameters will be taken as 2.5, 4.56, and 0.94 milliseconds, respectively (assuming an amplifier band-pass of 35 to 170 cps).

The response of the detector-amplifier to a ramp function of K v/sec slope and τ_4 seconds duration is

$$E_{out} = \frac{K}{\tau_1 \tau_3}\left[\frac{1}{s\left(s+\frac{1}{\tau_1}\right)\left(s+\frac{1}{\tau_2}\right)\left(s+\frac{1}{\tau_3}\right)}\right]\left[1 - e^{-\tau_4 s}\right] \qquad (5\text{-}22)$$

The corresponding time function, readily obtained by partial fraction expansion, is

$$E_{out(t)} = \frac{K}{\tau_1 \tau_3}\left[\frac{e^{-t/\tau_1}}{\left(\frac{1}{\tau_2}-\frac{1}{\tau_1}\right)\left(\frac{1}{\tau_3}-\frac{1}{\tau_1}\right)\left(-\frac{1}{\tau_1}\right)} + \frac{e^{-t/\tau_2}}{\left(\frac{1}{\tau_1}-\frac{1}{\tau_2}\right)\left(\frac{1}{\tau_3}-\frac{1}{\tau_2}\right)\left(-\frac{1}{\tau_2}\right)} + \frac{e^{-t/\tau_3}}{\left(\frac{1}{\tau_1}-\frac{1}{\tau_3}\right)\left(\frac{1}{\tau_2}-\frac{1}{\tau_3}\right)\left(-\frac{1}{\tau_3}\right)} + \tau_1 \tau_2 \tau_3\right]$$

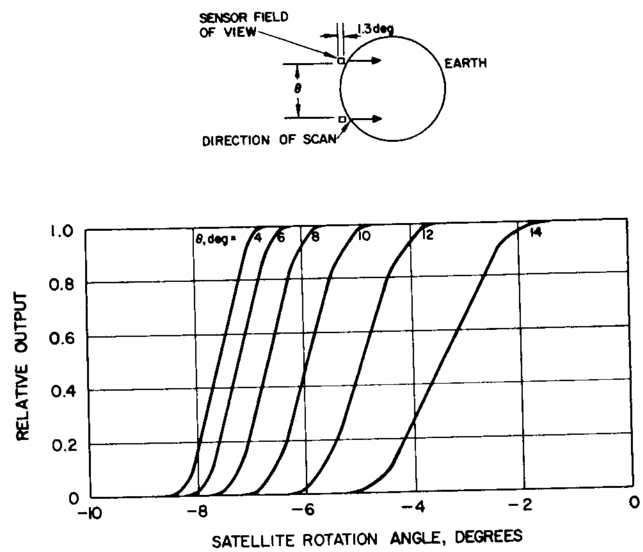


Figure 5-11. Earth Sensor Response Sensitivity for Different Values of θ

$$- \frac{K}{\tau_1 \tau_3} \left[\frac{e^{-\frac{t - \tau_4}{\tau_1}}}{\left(\frac{1}{\tau_2} - \frac{1}{\tau_1}\right)\left(\frac{1}{\tau_3} - \frac{1}{\tau_1}\right)\left(- \frac{1}{\tau_1}\right)} + \frac{e^{-\frac{t - \tau_4}{\tau_2}}}{\left(\frac{1}{\tau_1} - \frac{1}{\tau_2}\right)\left(\frac{1}{\tau_3} - \frac{1}{\tau_2}\right)\left(\frac{1}{\tau_2}\right)} \right.$$

$$\left. + \frac{e^{-\frac{t - \tau_4}{\tau_1}}}{\left(\frac{1}{\tau_1} - \frac{1}{\tau_3}\right)\left(\frac{1}{\tau_2} - \frac{1}{\tau_3}\right)\left(- \frac{1}{\tau_3}\right)} + \tau_1 \tau_2 \tau_3 \right]_{t \gtreqless \tau_4} \qquad (5\text{-}23)$$

This time function is plotted in Figure 5-12 for ramp function 5 of Figure 5-11 at a nominal spin rate of 100 rpm. The leading edge of the pulse may be used to trigger a threshold circuit set at an appropriate level.

The signal processing applied to the detector-amplifier pulses is illustrated in Figure 5-13. Threshold levels are set in two monostable multivibrators triggered by the detector-amplifiers. The outputs of these multivibrators are equal and opposite in magnitude and may be determined by zener diodes. If the outputs of the two multivibrators are summed together, a rectangular pulse is produced, the polarity of which indicates the time difference between the two sensor outputs. After a predetermined time period (longer than the time it takes for the field of view to scan the whole earth) the first multivibrator to be triggered reverts to its initial state, and in doing so causes the other multivibrator to also revert to its initial state.

The pulse train thus produced may be low pass filtered and the subsequent dc level telemetered on a narrow-band channel.

If the spin axis is misaligned such that the earth is not intersected by the sensors, no output will occur. This may be distinguished from the null position when the spin axis is properly aligned by the fact that a small precession of the spin axis will not cause an output to occur. If the sun or moon is intersected, maximum output will occur, but due only to a single sensor since the subtended angle (1/2 degree) is less than the angular separation of the two sensors.

If only one sensor intersects the earth, maximum output occurs, the polarity of which indicates which direction to precess the spin axis. Situations could arise in which one sensor intersects the earth, the other the sun or moon, etc., but these conditions are predictable and none of them gives the same as the desired output pattern near the null.

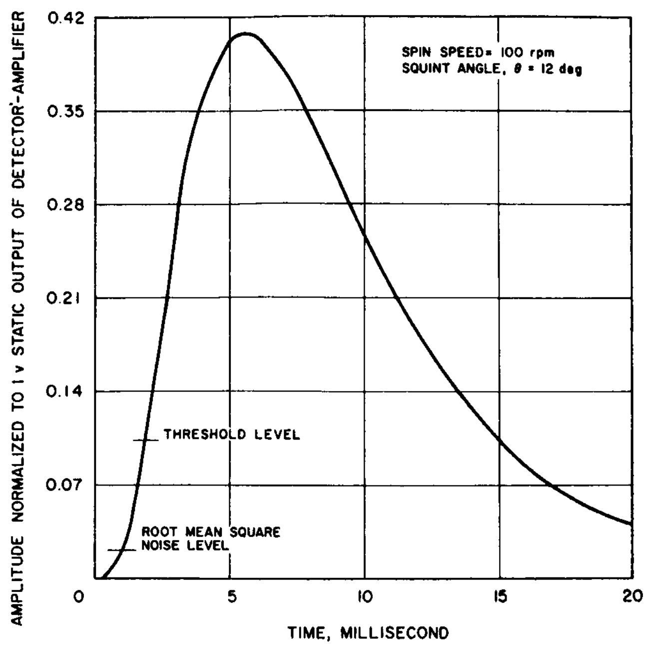


Figure 5-12. Detector Amplifier Output

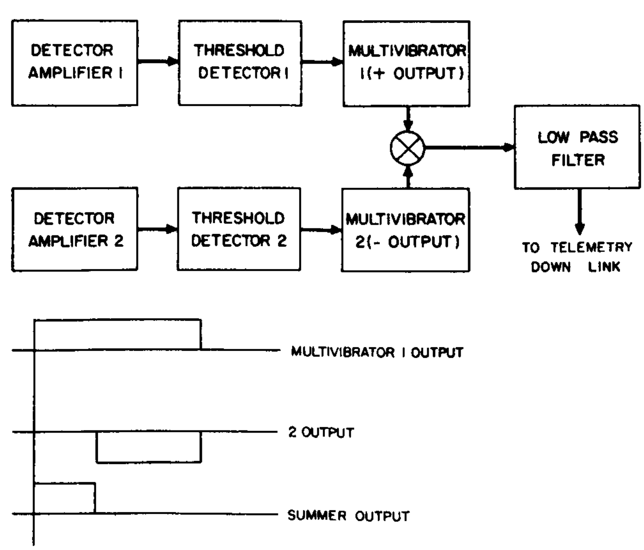


Figure 5-13. Signal Processing Logic
Block Diagram

The sources of error affecting the pointing accuracy are cell and amplifier noise, telemetry link noise, nonuniformity of infrared emission from earth, and boresight misalignment of the optical axis. Since, as previously stated, in the 14- to 18-micron radiation range the earth is a reasonably uniform source, that contribution of error may be taken as negligibly small, especially for a system utilizing edge detection.

In any consideration of the effect of detector-amplifier noise on the signal processing scheme outlined above, it is also necessary to consider noise triggering of the threshold detectors. If the noise output of the detector-amplifier is Gaussian, it may be shown that the instantaneous probability of

$$0 < V_{DA} < K\sigma = \frac{1}{2}\,\mathrm{erf}\,\frac{K}{\sqrt{2}} \tag{5-24}$$

For moderately large values of K

$$\frac{1}{2}\,\mathrm{erf}\,\frac{K}{\sqrt{2}} \cong \frac{1}{2}\left\{1 - \frac{e^{-K^2/2}}{K\sqrt{\pi/2}}\right\} \tag{5-25}$$

The instantaneous probability of

$$V_{DA} > K = 1 - \frac{1}{2} - \frac{1}{2}\,\mathrm{erf}\,\frac{K}{\sqrt{2}} = \frac{e^{-K^2/2}\cdot\sqrt{2}}{K\sqrt{\pi\cdot 2}} \tag{5-26}$$

$$V_{DA} > K = \frac{e^{-K^2/2}}{K\sqrt{2\pi}} \tag{5-27}$$

This is the probability that the threshold will be exceeded in a given sample space interval, which is determined by the amplifier bandwidth. This will be taken as the Nyquist interval (1/2B).

A reasonable false alarm rate might be taken as one per hour. Since there are two independent threshold detectors, either of which could cause a false alarm, there are 2 x 3600 x 270 = 1,944,000 independently contributing sample spaces in 1 hour for a detector-amplifier bandwidth of 135 cps. This sets the false alarm rate per threshold detector at 1/1,944,000 or 0.5015×10^{-6}, which in turn calls for a threshold at 4.96, according to Equation 5-19.

An estimation (Reference 5-9) of the combined cell and amplifier noise places the rms level at 0.02 of the static (dc) output or about 0.05 of the peak dynamic output of the detector amplifier. (This value appears to be lower than the previous analysis indicates, but is of no great concern since there is ample margin to raise the threshold for the same false alarm rate.) At the threshold level, the slope of the detector-amplifier pulse is 0.125 v/ms. The uncertainty of when the threshold circuit is triggered is then 0.02/0.125 = 0.16 millisecond as the variance or 0.226 millisecond as the variance of the difference between the two detector-amplifiers. This variance is reduced by two factors before entering the telemetry down link: by low-pass filtering of the multivibrator pulse and by the geometrical factor developed previously.

The bandwidth of the detector-amplifier is approximately 100 cycles wide. Since the spacecraft axis is not precessing rapidly, a 1 cps low-pass filter may be used, with a ten to one reduction of the time uncertainty. The time uncertainty is then 0.0226 millisecond or 0.0136 degree at a 100 rpm spin rate.

As the angular separation of the optic axes increases, the geometry further reduces the angular uncertainty, but this improvement is partially offset by decreasing attack angle of the limb by the sensor field of view, which reduces the slope of the detector-amplifier pulse, correspondingly increasing the time uncertainty.

A reasonable separation is taken as 13 degrees. The improvement factor is then 0.434 as computed from Equation 5-20. The transmitted angular pointing error is then 0.0059 degree. This very small error will be degraded by the telemetry down link if transmitted over a single telemetry channel because of the large dynamic range of the signal. The largest signal transmitted occurs when only a single sensor subtends the earth as is equivalent to ±18 degrees. If the peak signal-to-noise ratio of the telemetry channel is 30 db, the rms noise error is about 1.1 degrees, which is excessive. If a second telemetry channel is scaled to 35 times the sensitivity of the first, the 3σ variance of the noise angle is 0.1 degree. Therefore, the desirability of using two telemetry channels is evident for course and fine positioning of the spin axis.

The characteristics of the horizon scanner are as follows:

Dimensions, approximate	1.5 by 3 by 5 inches
Weight	0.8 pound
Detector	Immersed thermistor, τ = 2.5 milliseconds
Lens	Silicon
Effective focal ratio	f 0.21
Spectral bandpass	14 to 18 microns

Ambient temperature range	-10 to +60°C
Power consumption	200 milliwatts
Estimated angular pointing error of line from center of earth normal to spin axis at telemetry ground terminals (3σ)	0.1 degree

HORIZON SCANNERS (OGO)

The horizon scanner system consists of four independent scanner heads that individually search for, locate, and track the earth's horizon. Figure 5-14 shows the scan planes and tracking angles. In essence there are two operational phases of these scanners; in the search phase the scanners sweep through a fixed field of view in the scan plane, searching for an earth-space gradient to track. Physical design considerations limit the maximum scan angle of one scanner to approximately 90 degrees. In the track phase, the scan head tracks the gradient between earth and space. Four scanner heads are mounted in a crucifix form about the positive yaw axis of the vehicle so that the scan planes are 90 degrees apart and intersect the yaw axis. Thus, in the search phase the scanner system has a look angle of approximately 180 spherical degrees as long as the vehicle rotates rapidly about the yaw axis, and 360 degrees if the vehicle rotates rapidly about the pitch or roll axis.

Assembly Operation

Each tracker is independently capable of automatically searching over the complete 90 degrees scan range until the earth's horizon appears. The tracker will then lock on to the earth's horizon and revert to the track mode. In the track mode each tracker generates a signal that is directly proportional to the angle between the median scan direction and the direction of the horizon; this angle is defined as ϕ in Figure 5-15. The four trackers are defined by the letters A, B, C, and D in Figure 5-16. When all four trackers are tracking the horizon, four angular measurements are available for determining pitch and roll attitude. Only three are required, the fourth being available as a redundant capability. The angles that define pitch and roll error signals are given as follows (where a, b, c, and d are the outputs of scanners A, B, C, and D respectively):

Preferred Error Signals:

Pitch (ϵ_θ)	Roll (ϵ_ϕ)
$\frac{b - d}{2}$	$a - \frac{(b + d)}{2}$

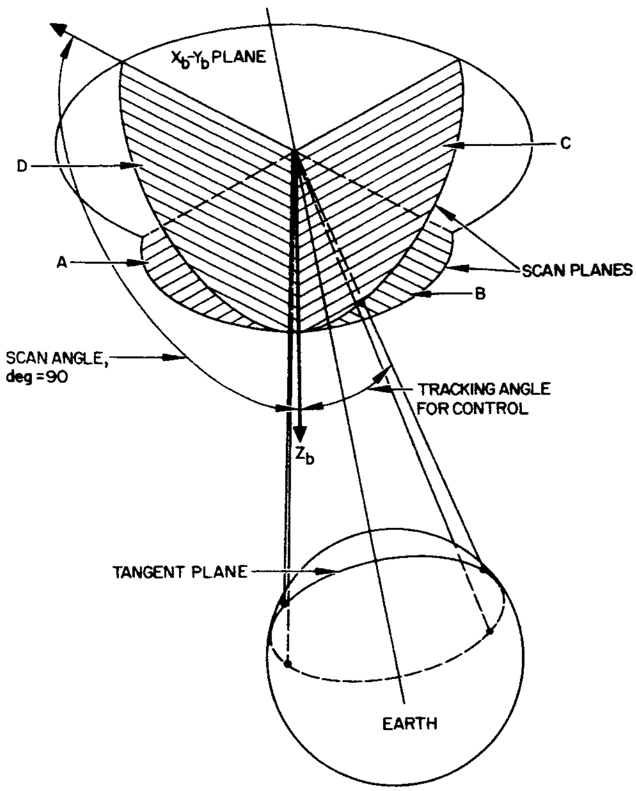


Figure 5-14. Illustration of Horizon Scanner Scan Planes and Tracking Angles

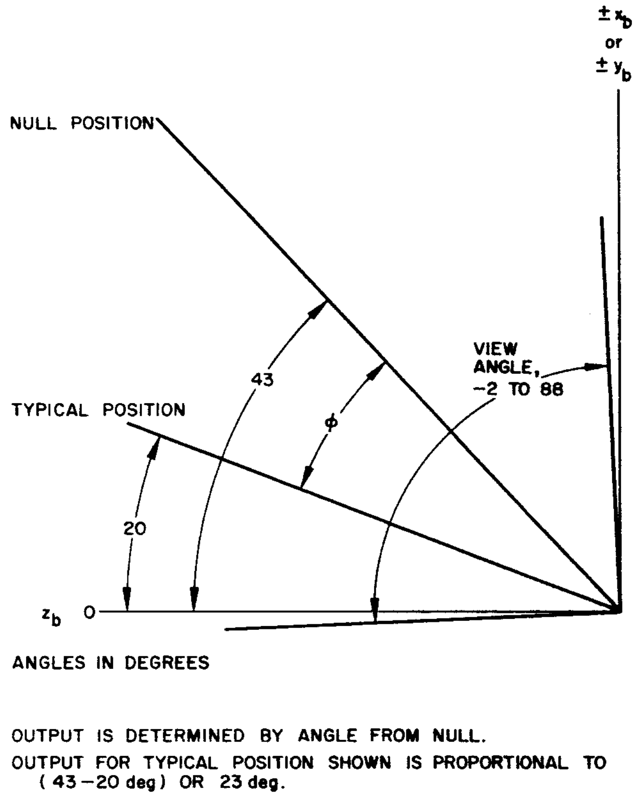


Figure 5-15. Scanner Head Geometry

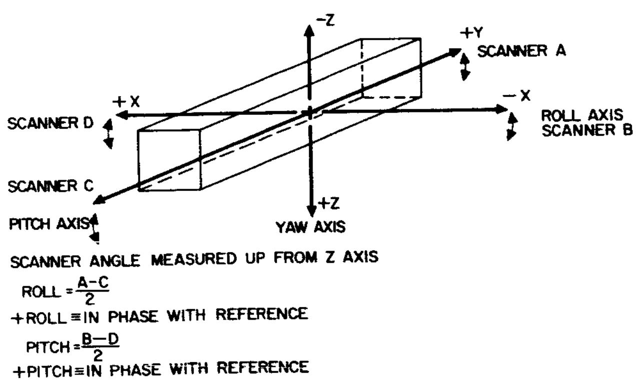


Figure 5-16. Coordinate System for OGO Horizon Scanner Assembly

Secondary positions (The subsystem logic selects the appropriate error signals):

ϵ_θ	ϵ_ϕ	Tracking Channel Failure
$\frac{b - d}{2}$	$\frac{(b + d)}{2} - c$	A
$\frac{a + c}{2} - d$	$\frac{(a - c)}{2}$	B
$b - \frac{(a + c)}{2}$	$\frac{(a - c)}{2}$	D

The above signals do not, of course represent linear pitch and roll error signals over their entire range. Nonetheless they provide quite satisfactory operation, even when limited at ϵ_ϕ = 25 degrees and ϵ_θ = 25 degrees.

System Description

Figure 5-17 is a block diagram of a single tracking head in the ATL horizon scanner assembly. The output of the Schmitt trigger goes to the drive amplifier which is, in essence, an integrator. This in turn drives the positor where the positor position is proportional to the current from the drive amplifier. Thus in the search mode, where there is no feedback through the optics, the Schmitt trigger will drive the positor in one direction until it hits the end of its search range at which time the zener diode will conduct and the resultant pulse will change the state of the Schmitt trigger. The trigger will then drive the drive amplifier in the opposite direction until once again the end of the search range is reached and the zener diode will conduct, changing the state of the Schmitt trigger. The search rate is approximately 100 degrees/second, more than satisfactory for the OGO operation.

Assuming that the scanning system is in search, as soon as the edge of the earth is detected (if the phase is proper) the state of the Schmitt trigger will be changed, and the positor will begin to oscillate the line of sight about the edge of the earth at the frequency determined by the servo loop, namely 13 cps, and at an amplitude (also so determined) of 1.6 degrees peak to peak.

Positor

The design is based on the use of the positor drive for scan motion. This drive utilizes a flexure pivot suspension to provide motion without sliding surfaces or friction. Basically, the scanning functions are accomplished by a mirror mounted on the rotor of a permanent magnet torquer, as shown in Figure 5-18. Also connected to the rotor are two coils that can move in a cylindrical air gap. A pair of flexure pivots connects the rotor to the base structure. Each pivot consists of a set of two flat springs attached to the rotor and to the base in such a way as to form an x. The flat sides of

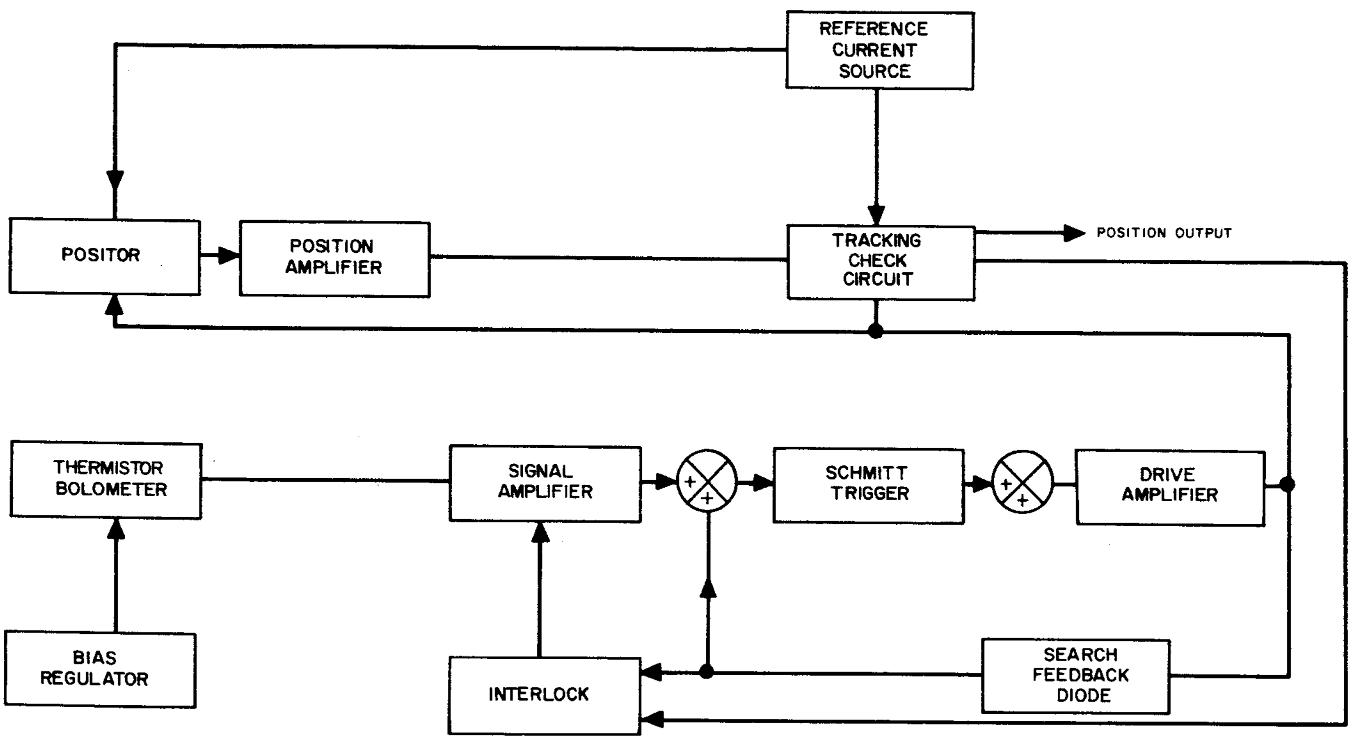


Figure 5-17. Horizon Scanner Block Diagram

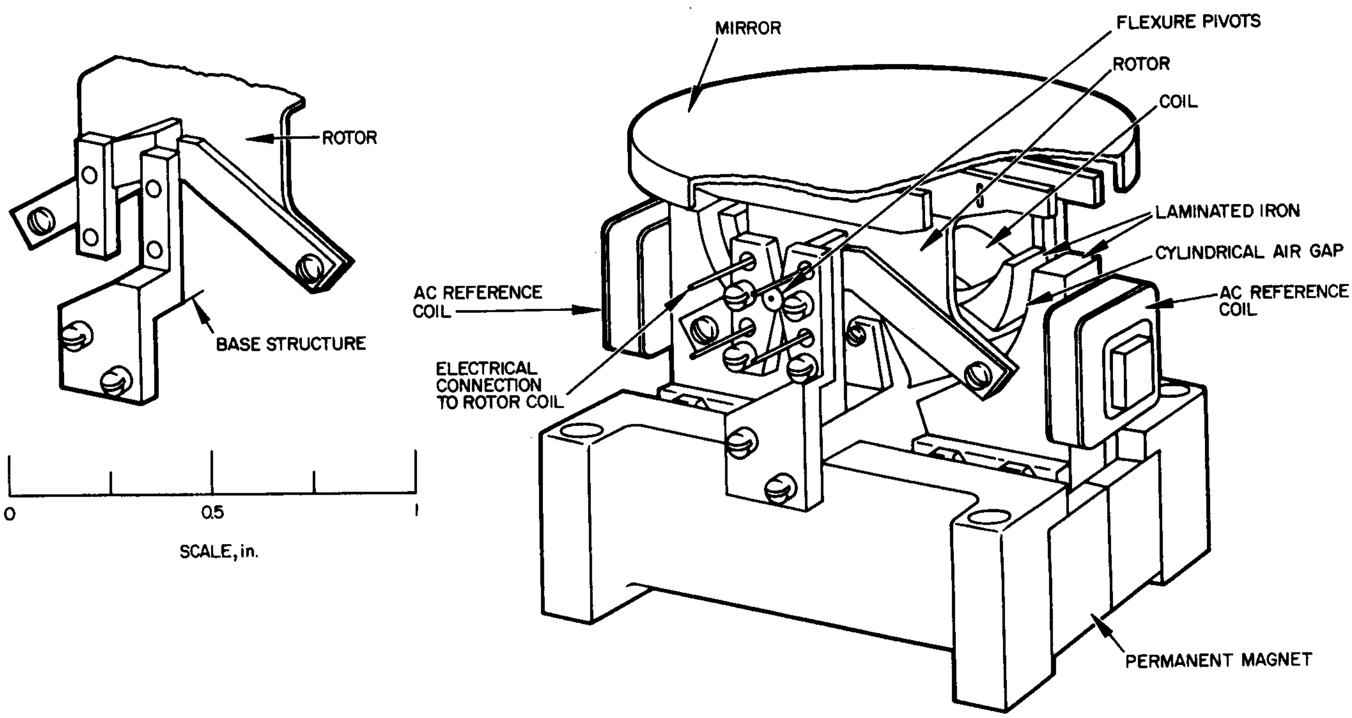


Figure 5-18. Positor

the springs are perpendicular to the point of the x, and the rotor turns about an axis perpendicular to the x, passing through its center point. The electrical connections to the rotor coils are made through the flexure pivots. The base structure contains a permanent magnet, a laminated iron magnetic structure to form the air gap, and two ac reference coils.

If a dc current is applied to the rotor coils, the rotor will assume an angular position proportional to the current. The mirror can be made to scan about some desired angle by applying a direct current plus a varying current to the rotor coils. The total angular motion of the positor is ±22.5 degrees, resulting in a total optical range of ±45 degrees.

The ac reference coils set up a small amplitude, high frequency (2461 cps) flux in the air gap. The high frequency signal with an amplitude proportional to the angular position of the rotor from the mechanical null position is therefore present in the rotor coils. An accurate, linear indication of position is then obtained which is independent of the flexure pivot spring rate.

Optics

An optical schematic of a tracker is shown in Figure 5-19. Incident radiation is reflected by a positor-driven plane mirror to a telescope, which consists only of a simple germanium objective lens with a germanium immersed thermistor bolometer in the focal plane. The positor drive makes possible the use of such simple optics. The mirror is placed in front of the telescope to eliminate all possibility of trouble due to background modulation and unwanted edge glint.

The telescope is mounted in such a manner that when the system is oriented the scanning mirror is about an axis along the horizon; therefore, the line of sight moves above and below the horizon. In Figure 5-19 this axis is perpendicular to the plane of the paper. During search, the mirror causes a line of sight to traverse back and forth over the complete 90-degree range at a linear rate of about 100 degrees/second. During track, the line of sight varies sinusoidally about the horizon with approximately ±0.8 degree amplitude and a 13 cps frequency.

Use of this small amplitude scan pattern in tracking makes the attainment of the required accuracy easier, as well as the attainment of sufficient electrical and mechanical resolution and drift characteristics. Also, only a signal point on the horizon is scanned by any tracker, so that little variation in target temperature would be expected within the scan cycle.

Sun Protection

The system is so designed that tracking the sun would not result in any damage to the scanner.

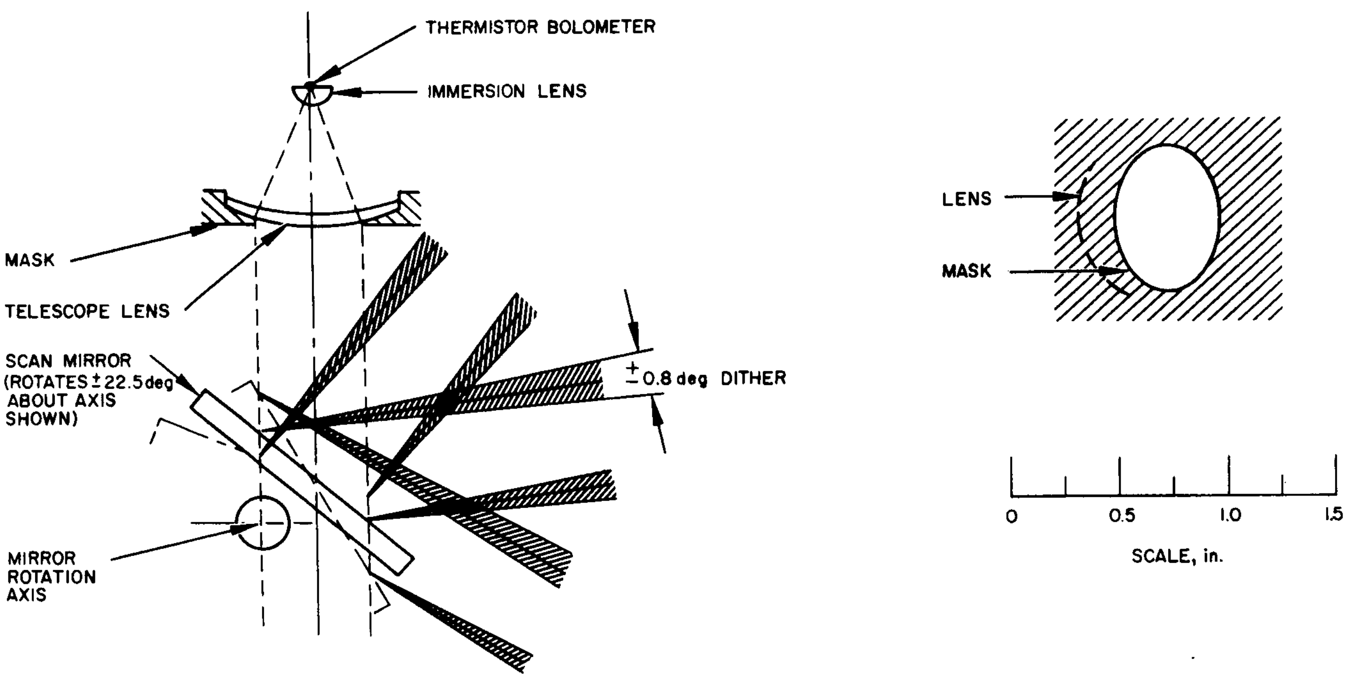


Figure 5-19. Optical Schematic of Tracker

Electronics

The tracker electronics consists of three sections: 1) drive, 2) position readout, and 3) tracking check. Each will be discussed separately.

Drive Section. The drive section consists of thermistor bolometer with its bias regulator, the signal amplifier, a Schmitt trigger, a positor drive amplifier, and the search feedback diode. The bias regulator supplies the proper bias current to the bolometer, using a combination of active and passive filters. The output signal from the bolometer is amplified in the signal amplifier and applied to the input of the Schmitt trigger. This trigger is biased so as to remain stable in either state as long as no input is applied. The trigger output switch is between ±10 volts and is applied directly to the drive amplifier. Frequently selective negative feedback is applied to the drive amplifier to obtain the required transfer function. The drive amplifier output furnishes the drive coil current to move the positor.

Since gradients other than the horizon gradient might be present within the earth, it is desirable to permit this transition from search to track to occur only on the first gradient encountered while searching from space towards the earth. This feature is provided by an interlock which, when off, prevents the signal amplifier from triggering the Schmitt. The interlock is turned off whenever search action is initiated, and can be turned on only by a pulse from the search feedback diode, occurring at the upper search limit.

Position Readout Section. The position readout section consists of the reference current generator, the position amplifier, and two switches activated by the tracking check signal. As described previously, a 2461 cps voltage will be introduced in the drive coils, which will be linearly dependent upon the displacement of the positor from its null position. This position readout signal is separated from the drive current by a parallel tuned circuit in series with the drive coils and amplified by the position amplifier.

Tracking Check Section. The tracking check circuit is another Schmitt trigger, with its inputs arranged so that it will be in the tracking state if the signal amplifier output is above a specified value, indicating that either the earth or sun is being tracked. Discrimination between earth and sun is provided by a sun-alarm circuit, which is triggered to its sun-alarm state when the sun appears in the field of view of a tracker head.

REFERENCES

5-1. Syncom II Summary Report, Hughes SSD 3128R, NASA Contract 5-2797, 31 March 1963.

5-2. Otten, D. D., OGO Attitude Control Subsystem Description, Logic, and Specifications," STL 2313-004-RU-000, 4 December 1961.

5-3. Blakemore, D. J., "Pitch Rate for Earth Acquisition," STL 9313.8-96, 7 August 1961.

5-4. "Communication Satellite Project Advent, Summary Report on The Preliminary Design Inspection," General Electric MSVD, Doc. No. 61SD4262, 28 August 1961.

5-5. "Proceedings of the International Meteorlogical Satellite Workshop," Washington, D. C., Nov. 13-22, 1961.

5-6. "Radially Oriented Horizon Sensor," Barnes Engineering Company Specification Sheet 13-200.

5-7. Barnes Engineering Company, Infrared Bulletin 14-002 (Two reprints from "Electronics," September 22 and 29, 1961, issues).

5-8. "Infrared Instrumentation for Meteorlogical Satellites," Barnes Engineering Company, Bulletin 14-003.

5-9. Barnes Engineering Company (verbal).

6. SPACECRAFT SYSTEMS DESIGN

SPACECRAFT SUBSYSTEMS PERFORMANCE REQUIREMENTS SPECIFICATION AND BLOCK DIAGRAM

1.0 INTRODUCTION

1.1 Purpose: The purpose of this specification is to define requirements to which each subsystem of the Syncom II is to be designed and tested.

1.2 Scope: This specification defines what is required of each subsystem of the Syncom II.

2.0 APPLICABLE DOCUMENTS

2.1 The following documents form a part of this specification to the extent specified herein:

MIL-W-8160D	Dated 17 March 1961 Wiring, Guided Missile Installation of General Specification for
MIL-I-26600(USAF)	Dated 2 June 1958 Amendment 1, dated 17 June 1959 Interference Control Requirements Aeronautical Equipment
General Range Safety Plan Volume I, Missile Handling	Dated 1 April 1960, Errata Sheet Dated 4 May 1960, Revision 1 Dated July 1960, Revision 2
LMSC-A057612	Dated 30 September 1962 Syncom Booster Feasibility Study Final Design Report Lockheed Missile and Space Company

Technical Memorandum 732	Dated October 1962 Environment of Syncom Mark II Paul M. Blair, Jr. and Herbert T. Toda
S2-0100	Dated 18 February 1962 Performance and Test Specification Advanced Syncom Spacecraft
	Dated 15 May 1963 Syncom II RF and Electrical Interface Specification
	Dated 15 May 1963 Syncom II Mechanical Interface Specification
NASA Document MSFC-PROC-158B	Dated 15 February 1963 Procedure for Soldering of Electrical Connectors

3.0 REQUIREMENTS

3.1 Definition of Spacecraft Subsystems: The major and minor control items have been grouped together into functional groups as subsystems. These subsystems and the control items of which they are composed are listed below and shown diagrammatically in Figures 6-1 through 6-4.

1) Communication Subsystem
475025, 475030, 475040

2) Antenna and Jet Control Subsystem
475035, 475303, 475160

3) Telemetry and Command Subsystem
475045, 475050, 475055

4) Power Supply Subsystem
475060, 475251, 475252, 475253, Battery

5) Spacecraft Structure Subsystem
475065, 475301, 475302, 475304, Separation switch

6) Wire Harness Subsystem
475300

Figure 6-1. Syncom Block Diagram 1

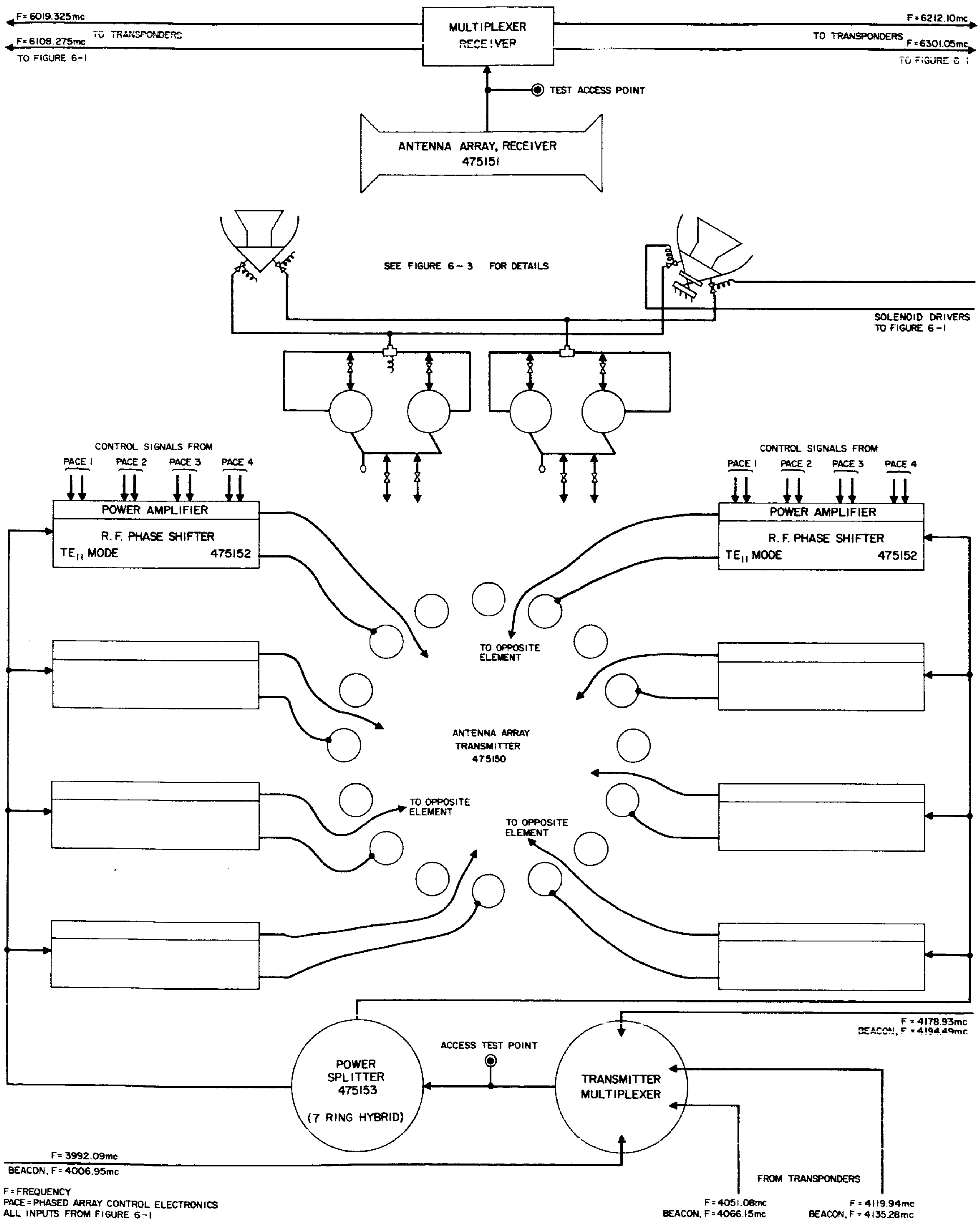


Figure 6-2. Syncom Block Diagram 2

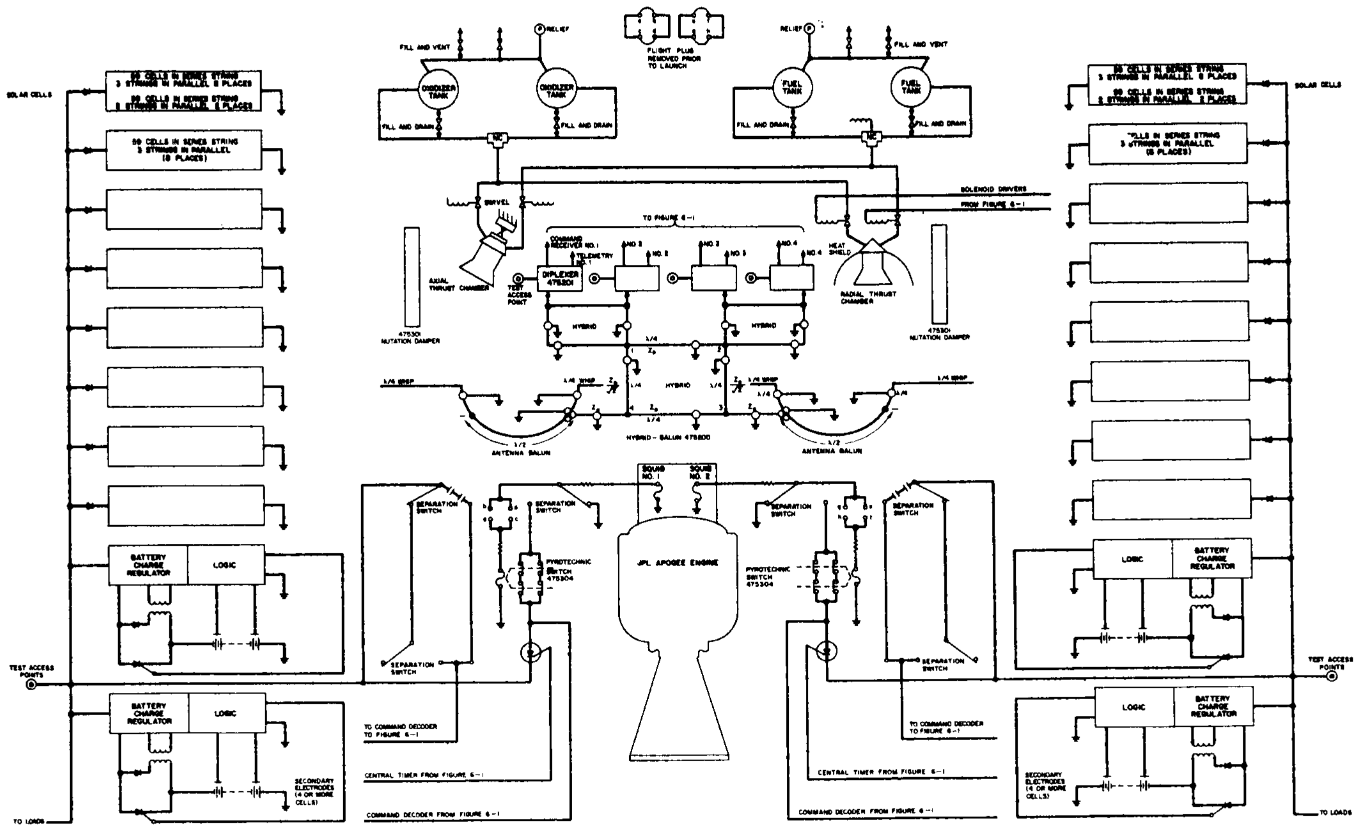


Figure 6-3. Syncom Block Diagram 3

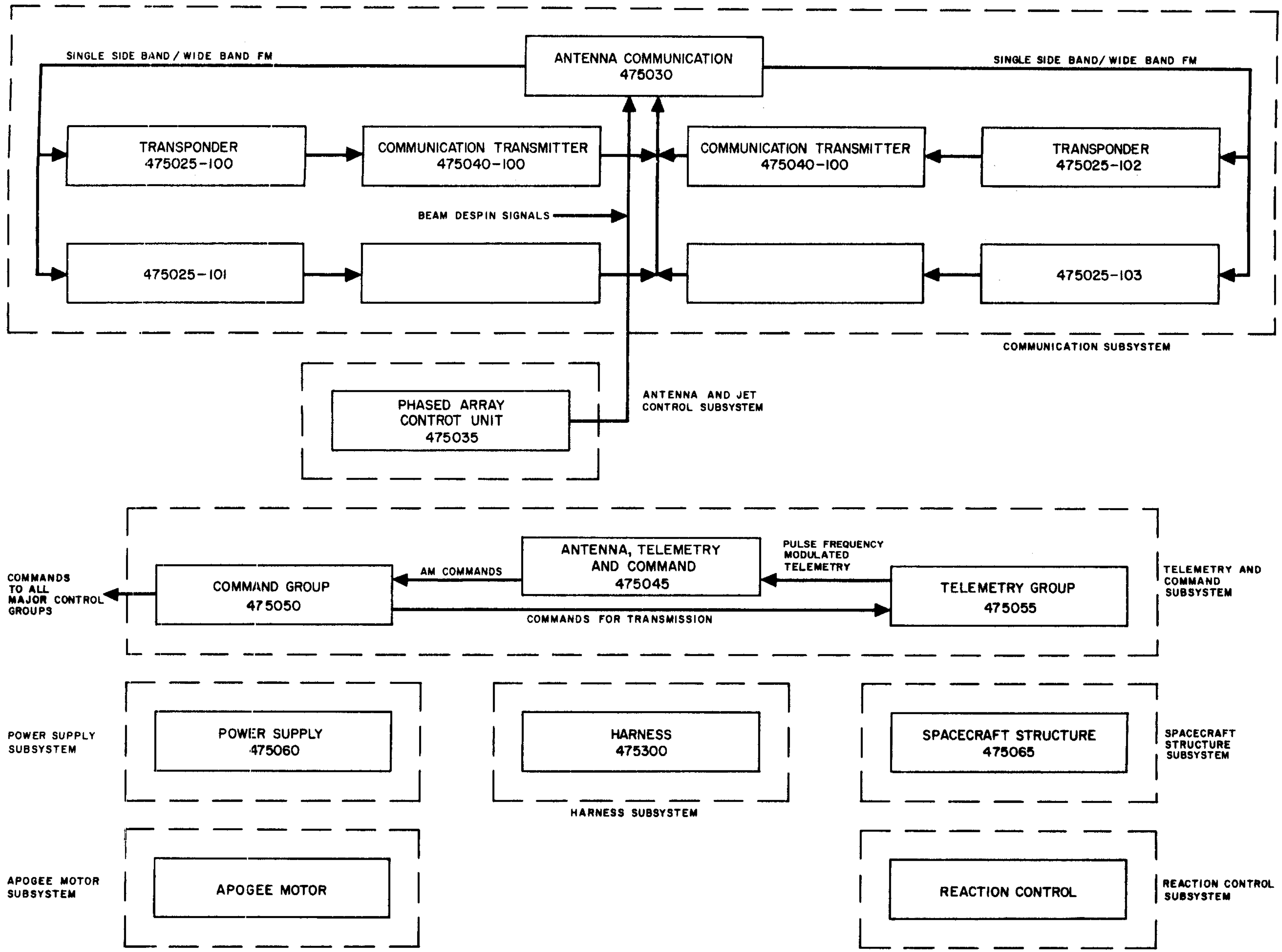


Figure 6-4. Subsystems and Major Control Items

7) Apogee Motor Subsystem

8) Reaction Control Subsystem

3.2 Communication Subsystem: The communication subsystem shall provide facilities to receive, convert frequency, amplify, and retransmit microwave signals. Each transponder shall be capable of operating in either a frequency-translation mode or a multiple-access mode. An unmodulated beacon signal shall be transmitted to provide a signal for ground antenna autotrack. Series and/or paralleled redundant units shall be used as necessary (consistent with weight and volume limitations) to satisfy reliability requirements. The communication subsystem shall be composed of:

1) Four Communication Transponders, 475025

2) One Communication Antenna, 475030

3) Four Communication Transmitters, 475040

3.2.1 Reliability: The communication subsystem shall have a probability of operation within the performance requirements of

0.906 for a 1-year requirement
0.525 for a 3-year requirement

3.2.2 Communication Transponder, 475025

3.2.2.1 Quantity: There shall be four communication transponders.

3.2.2.2 Modes: Each transponder shall be capable of operating in either the frequency-translation or the multiple-access mode.

3.2.2.3 Frequency Assignments: The frequency assignment for transponders shall be (in mc) as given in Table 6-1.

3.2.2.4 Common Requirements: Each mode of the transponder shall meet the following requirements.

3.2.2.4.1 Input and Output Impedance: The input and output impedance of each receiver of the transponder shall be approximately 50 ohms.

3.2.2.4.2 Noise Figure: The noise figure of each receiver shall be better than 9 db (referenced to the standard noise temperature of 290°K).

TABLE 6-1. TRANSPONDER FREQUENCY ASSIGNMENTS (MC)

Transponder	Input	Output	Beacon	Master Oscillator		IF Frequency Transponder
				Multiple Access	Frequency Transponder	
475025-100	6019.325	3992.09	4006.95	31.1882	15.83776	62.4
475025-101	6108.275	4051.08	4066.16	31.6491	16.01718	63.3
475025-102	6212.10	4119.94	4135.28	32.1870	16.34496	64.4
475025-103	6301.05	4178.93	4194.49	32.6979	16.57901	65.2

3.2.2.4.3 Power Out: Each receiver shall have a power out of 1 mw ± ____ mw.*

3.2.2.4.4 Telemetry Outputs: Each receiver shall provide an output from the IF strip for transmission by telemetry transmitter.

3.2.2.4.5 Transponder Power: Each transponder shall require no more than 75 ma at 24 volts.

3.2.2.5 Frequency Translation Receiver, Peculiar Requirements: The frequency translation receiver shall translate and amplify the signal carrier frequency with no conversion in modulation.

3.2.2.5.1 RF Bandwidth: The 3-db bandwidth for the frequency-translation receiver shall be 25 mc ±1.5 mc measured between IF input and RF output.

3.2.2.5.2 Receiver Carrier Power: The preceding requirements shall not be imposed on the receiver unless the received carrier power exceeds -101.2 dbw.

3.2.2.5.3 Received Noise Power: The preceding requirements shall not be imposed on the receiver unless the received noise power is less than -121.3 dbw.

3.2.2.6 Multiple-Access Receiver, Peculiar Requirements: The multiple-access receiver shall convert the single-sideband signals from the IF strip into a phase-modulated signal. This signal shall be multiplied up to the proper microwave frequency and amplified.

3.2.2.6.1 RF Bandwidth: The 3-db bandwidth of the multiple-access receiver shall be 6 mc (+1 mc, -0.5 mc) measured at the preamplifier output.

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*Certain parameters have been omitted because applicable data were not available at time of publication.

3.2.2.6.2 Phase Modulator Distortion Noise: The distortion noise generated in the phase modulator shall be so low that it does not represent a limiting factor in meeting CCIR's signal/noise recommendations for noise channels when companders are used.

3.2.2.6.3 Capacity: Each of the multiple-access receivers shall be capable of conveying up to 1200 one-way 4-kc voice channels.

3.2.2.6.4 Test Tone/Fluctuation Noise Ratio: This ratio shall be greater than 47.6 db.

3.2.2.6.5 Test Tone/Intermodulation Noise Ratio: This ratio shall be greater than 50.5 db.

3.2.2.6.6 Test Tone/Noise Ratio: This ratio shall be greater than 45.8 db.

3.2.2.6.7 Inputs-Outputs: The transponder inputs-outputs shall be as given in Table 6-2.

3.2.3 Communications Antenna - 475030: The antenna unit shall receive the incoming 6-gc signals, separate them into the four frequency channels, and supply them to the appropriate receiver. The antenna unit combines the four 4-gc signals from the transmitter unit, processes and transmits them.

3.2.3.1 Receiving Antenna: The receiving antenna shall be capable of receiving 6-gc signals.

3.2.3.1.1 RF Power In: None of the performance requirements of the communication subsystem shall apply unless the input power is at least -106.7 dbw.

3.2.3.1.2 Gain: The antenna gain shall be at least 8 db over the frequency range of 6019.325 to 6301.05 mc.

3.2.3.1.3 Receiving Antenna Pattern Characteristics: The radiation pattern of the receiving antenna shall be omnidirectional in the ϕ-plane and have a minimum beamwidth of 17.3 degrees in all θ planes. The peak of the beam will be at an angle of θ = 90 degrees in all directions of ϕ.

3.2.3.2 Receiver Multiplexer: The receiver multiplexer shall be used to separate the received 6-gc signals into the four frequency channels.

3.2.3.2.1 Input and Output Impedance: The input and output impedance shall be as close as is possible to 50 ohms.

TABLE 6-2. TRANSPONDER - 475025

Inputs and Outputs

Control Item	Number of Quadrants	Function	Description
INPUTS FROM:			
Multiplexer	1	Multiple access	AM/SSB
Multiplexer	1	Frequency translation	WBFM
475211	4	Turn on multiple-access series regulator	Command pulse
475211	4	Turn off multiple-access series regulator	Command pulse
475211	4	Turn on frequency translation series regulator	Command pulse
475211	4	Turn off frequency translation series regulator	Command pulse
OUTPUTS TO:			
475171	1	Multiple-access output	PM
475171	1	Frequency translation output	WBFM
475221	4	Multiple-access signal strength	
475221	4	Frequency translation signal strength	

The standard command pulse shall be 60 msec long, have a 5-msec rise time, and be 0.2 volt in amplitude.

3.2.3.2.2 Frequency: The multiplexer shall be capable of separating the input into four separate frequency channels:

1) 6019.325 mc

2) 6108.275 mc

3) 6212.10 mc

4) 6301.05 mc

3.2.3.2.3 Bandwidth: The bandwidth on the four frequencies listed above shall be ±12.5 mc.

3.2.3.2.4 Losses: The maximum loss shall be 1.04 db or less over the frequency range.

3.2.3.2.5 RF Power Input: The power in shall be at least -100.2 dbw.

3.2.3.2.6 Isolation: Isolation between frequency channels shall be at least 17 db at f_o ±44.8 mc (f_o denotes each of the four frequencies listed in 3.2.3.2.2).

3.2.3.3 Transmitting Multiplexer: The transmitting multiplexer shall be used to combine the four 4-gc signals.

3.2.3.3.1 Input and Output Impedance: The input and output impedance shall be as close as possible to 50 ohms.

3.2.3.3.2 Frequency: The transmitting multiplexer shall be capable of accepting frequencies from 3979.59 mc to 4194.49 mc.

3.2.3.3.3 Losses: The losses shall not exceed 0.7 db for the four transponder frequencies as listed in 3.2. 2.2. The beacon loss shall not exceed 2.0 db.

3.2.3.3.4 RF Power Inputs: The RF power input shall be at least 5.7 dbw.

3.2.3.3.5 Isolation: Isolation between channels shall be at least 17 db at f_o ±44.8 mc (f_o denotes each of the four frequencies listed in 3.2.3.2.2).

3.2.3.4 Phased Array Transmitting Antenna: For the purpose of specifying antenna gain and pattern characteristics, the phased array is defined as consisting of the 16 collinear arrays that make up the radiating portion of the antenna, the transmission lines leading from the outputs of the phase shifters to the element arrays and any matching networks required to obtain a broadband impedance match of the element arrays, the phase shifters, and eight-way power divider.

3.2.3.4.1 RF Power Splitter - 475153: The RF power splitter shall split the RF power into eight equal amplitude and equal phase parts.

3.2.3.4.1.1 Frequency: The frequency range shall be from 3992.09 mc to 4194.49 mc.

3.2.3.4.1.2 Losses: The losses shall be no more than 1 db over the frequency range.

3.2.3.4.1.3 RF Power Input: The RF power input shall be at least 5.0 dbw.

3.2.3.4.1.4 Input and Output Impedance: The input and output impedance shall be made as close as possible to 50 ohms.

3.2.3.4.2 Phase Shifter, 475152: The phase shifters shall be capable of inducing 16 different phase shifts (ϕ_n) of $\phi_n = 2\pi\cos(wt + n \cdot 22.5$ degrees), n = 0, 1, 2, ... 15, where the spacecraft spin rate, ω, shall be $200\pi \pm 100\pi$ radians.

3.2.3.4.2.1 Losses: The maximum losses shall be 1 db over the frequency range.

3.2.3.4.2.2 RF Power Input: The minimum RF power input shall vary over the range 0.4 w/phase shifter to 2 w/phase shifter.

3.2.3.4.2.3 Input and Output Impedance: The input and output impedance shall be made as close as possible to 50 ohms.

3.2.3.4.3 Phased-Array Transmitting Antenna, 475150

3.2.3.4.3.1 RF Power Input: The RF power input shall be at least 3 dbw.

3.2.3.4.3.2 Phased-Array Antenna Gain: The pattern gain, at the peak of the beam, shall be at least 18.0 db over the frequency band from 3992.09 mc to 4194.49 mc.

3.2.3.4.3.3 Phased-Array Antenna Pattern Characteristics: The radiation pattern of the transmitting antenna shall be an elliptically shaped pencil beam a minimum of 17.3 degrees wide in the θ-plane (parallel to the spin axis) and 23 degrees wide in the ϕ-plane (perpendicular to the spin axis). The peak of the beam will be at an angle $\theta = 90$ degrees for any direction of the beam in the ϕ-plane.

3.2.4 Communication Transmitter, 475040: The communication transmitter shall consist of a power amplifier, amplifier power supply, and input-output buffer equipment.

3.2.4.1 Quantity: There shall be four communication transmitters. Each communication transmitter shall consist of:

1)	One 3-db hybrid	475171
2)	One telemetry monitor	475177
3)	One RF switch	475173
4)	Two traveling-wave tubes	384H
5)	Two TWT power supplies	475174

3.2.4.2 3-db Hybrid, 475171: The 3-db hybrid shall be capable of accepting signals from two separate sources and coupling them to either of two separate outputs.

3.2.4.2.1 Losses: The losses including the power split shall not exceed 3.25 db over the frequency range, 3992.09 mc to 4194.49 mc.

3.2.4.2.2 RF Power Input: The RF power input shall be 1 mw ±_____ mw.

3.2.4.2.3 Isolation: Isolation shall be at least 25 db.

3.2.4.2.4 Input and Output Impedance: Input and output impedance shall be made as close as possible to 50 ohms.

3.2.4.2.5 Input-Outputs: 3-db hybrid inputs-outputs shall be as given in Table 6-3.

3.2.4.3 Power Amplifiers: The final power amplifiers shall be traveling-wave tubes 384H.

3.2.4.3.1 Input Power: The input power shall be 1/2 mw ±_____ mw.

3.2.4.3.2 Output Power: The RF output power of each power amplifier shall be at least 4 watts.

3.2.4.3.3 Frequency Band: The TWT shall be capable of operating within specifications over the frequency range 3992.09 mc to 4194.49 mc.

TABLE 6-3. HYBRID - 475171

Inputs and Outputs

Control Item	Number of Quadrants	Function	Description
INPUTS FROM:			
475025	1	Multiple access	PM, 1 mw
475025	1	Frequency translation	WBFM, 1 mw
OUTPUTS TO:			
384H	1	Multiple access	PM, 1/2 mw
384H	1	Frequency translation	WBFM, 1/2 mw

3.2.4.3.4 TWT Electrical Performance: TWT electrical performance shall be found in Table 6-4.

3.2.4.3.5 Inputs-Outputs: TWT inputs-outputs shall be found in Table 6-5.

3.2.4.4 TWT Power Supply, 475174: The power supply shall provide necessary power for TWT operation.

3.2.4.4.1 Power Input: The power input shall be -24 volts ±1 percent.

3.2.4.4.2 Cathode Voltage: The cathode voltage shall be -1300 volts ± __ percent.

3.2.4.4.3 Helix Voltage: The helix voltage shall be 0 volt.

3.2.4.4.4 Collector Voltage: The collector voltage shall be -725 volts ± __ percent.

3.2.4.4.5 Anode Voltage: The anode voltages shall be 125 volts ± __ percent.

3.2.4.4.6 High Voltage Start: There shall be a high voltage start pulse. The start pulse shall be a standard command pulse.

3.2.4.4.7 Filament Voltage: The filament power supply shall be -4.5 volts ac ± __ percent.

TABLE 6-4. ELECTRICAL PERFORMANCE - TWT 384

Frequency	3.9 gc - 4.2 gc
RF power output	3.9 watts - 4.3 watts
RF saturation gain	37.2 db
RF small signal gain	50 db
Spurious output (harmonics of operating frequency)	
Noise figure	28 db
Impedance	50
VSWR (input and output)	1.2:1
Maximum load VSWR	Short circuit, any phase
Intermodulation distortion	
Efficiency (excluding heater)	35 percent
Heater power	1.17 watts nominal
Total dc input power	12.1 watts
Cathode voltage	-1300 volts ± ___ percent
Collector voltage	-725 volts ± ___ percent
Collector current	17.6 ma
Helix voltage	0 volt
Helix current	1.7 ma
Anode voltage	125 volts ± ___ percent
Anode current	0
Heater voltage	4.5 volts ± ___ percent
Heater current	0.27 ampere

TABLE 6-4 (continued)

Predicted life	50,000 hours
Focusing	Platinum - Cobalt magnets Field strength - 750 gauss
Beam transmission with RF	85.5 percent
Cathode	
Base	Ni
Impurities	0.09 percent Zr 0.02 percent Fe 0.001 percent Mn 0.001 percent Si 0.02 percent Cu 0.005 percent W
Cathode loading	0.0854 amp/cm^2
Cathode temperature	720°C

TABLE 6-5. TRAVELING-WAVE TUBE 384-H

Inputs and Outputs

Control Item	Number of Quadrants	Function	Description	
INPUT FROM:				
475174	1	Cathode	-1300 volts	19.3 ma
475174		Helix	0 volt	1.7 ma
475174	1	Anode	125 volts	0 ma
475174	1	Collector	-725 volts	17.6 ma
475174	1	Filament	4.5	0.27 amp
475171	1	RF	4 gc	1/2 mw
OUTPUT TO:				
475173	1	RF	4 gc	4 watts

3.2.4.4.8 Transmitter Regulators: The transmitter regulators shall be a series type.

3.2.4.4.8.1 Power Input: The power input shall be -28 volts unregulated.

3.2.4.4.8.2 Turn-on/Turn-off: The transmitter regulators shall be capable of being turned on and turned off by a standard command pulse from any of the three regulators.

3.2.4.5 Inputs-Outputs: Transmitter regulator inputs-outputs shall be as given in Table 6-6.

3.2.4.6 RF Switch, 475173: The RF switch shall be capable of switching signals from either of two inputs to one output.

3.2.4.6.1 RF Power Input: The RF power input shall be at least 4 watts.

3.2.4.6.2 Frequency Band: The RF switch shall be capable of operating within specification over the frequency range, 3992.09 mc to 4194.49 mc.

3.2.4.6.3 Losses: The losses shall not exceed 0.3 db over the frequency range 3992.09 mc - 4194.49 mc.

3.2.4.6.4 Switching Power: The switching power shall not exceed 1.0 ampere.

3.2.4.6.5 Input and Output Impedance: The input and output impedance shall be made as close as possible to 50 ohms.

3.2.4.6.6 Inputs-Outputs: RF switch inputs-outputs shall be as given in Table 6-7.

3.2.4.7 Telemetry Monitor, 475172: The telemetry monitor shall be capable of monitoring a 4-watt signal for telemetry purposes.

3.3 Antenna and Jet Control Subsystem: The antenna and jet control subsystem is comprised of four sets each of three control items:

1) Phased-Array and Jet Control Electronics, 475035

2) Central Timer, 475303

3) Series Regulator 475160

TABLE 6-6. POWER REGULATOR TRAVELING-WAVE TUBE
Inputs and Outputs

Control Item	Number of Quadrants	Function	Description	
INPUTS FROM:				
475211	4	Turn on	Command pulse*	
475211	4	Turn off	Command pulse	
475211	4	High–voltage start	Command pulse	
OUTPUTS TO:				
TWT 384H	1	Anode	+125 volts	0 ma
TWT 384H	1	Helix	0 volts	1.7 ma
TWT 384H	1	Cathode	-1300 volts	19.3 ma
TWT 384H	1	Collector	-725 volts	17.6 ma
TWT 384H	1	Filament	4.5 volts	0.27 amp
475173	1	Switch RF		1 amp

*Standard command pulse shall be 60 msec long, have a 5 msec rise time, and be 0.2 volt in amplitude.

This subsystem is responsible for firing the apogee motor at the proper time; developing control signals to provide the spacecraft with the capability of being properly oriented and synchronous in the equatorial plane; and maintaining antenna beam despin rate.

3.3.1 Reliability: The antenna and jet control subsystem shall have a probability of operation within the performance requirements of

0.988 for a 1-year requirement

0.963 for a 3-year requirement

TABLE 6-7. RF SWITCH - 475173
Inputs and Outputs

Control Item	Number of Quadrants	Function	Description
INPUTS FROM:			
TWT 384H No. 1	1	RF	4 gc 6 dbw
TWT 384H No. 2	1	RF	4 gc 6 dbw
475174 No. 1	1	Switch	1 ampere
475174 No. 2	1	Switch	1 ampere
OUTPUTS TO:			
4715154	1	RF	5.7 dbw

3.3.2 Phase-Array and Jet Control Electronics

3.3.2.1 Quantity: There shall be four phased-array and jet control electronics.

3.3.2.2 Phased-Array Control Electronics (PACE): Each PACE shall be able to operate independently of the other three PACE.

3.3.2.2.1 Signals Out: There shall be 16 signals out of the PACE. They shall be $20 \cos 2\pi\ [\sin (2\pi ft + m\pi/8)]$ and $20 \sin 2\pi\ [\sin (2\pi ft + m\pi/8)]$ where $m = 0, 1, 2, \ldots, 7$.

3.3.2.2.2 Error: The positioning error on the beam shall not exceed ± ___ degrees.

3.3.2.3 Jet Control Electronics: The jet control electronics shall be capable of producing four jet fire pulses.

3.3.2.3.1 Command Beam Angle: The command beam angle shall be the angle at which the jets fire. They shall be standard command pulses.

3.3.3 Central Timer, 475303

3.3.3.1 Quantity: There shall be four central timers.

3.3.3.2 Squib Fire Signals: The central timer shall provide a fire signal 315 minutes ± 1 percent after separation. Squib fire signal from at least two central timers shall be necessary to fire squibs.

3.3.3.3 Relative Motion Correction Pulse: The central timer shall provide 512 pulses per day.

3.3.4 PACE Inputs-Outputs: PACE inputs-outputs shall be as given in Table 6-8.

3.3.5 Series Regulator, 475160: Each PACE shall have a series regulator for a power supply.

3.3.5.1 Power Out: The regulator shall provide +24 volts at 200 ma and -24 volts at 100 ma. Regulation shall be ± 1 percent. Maximum ripple shall be 200 mv peak-to-peak.

3.3.5.2 Turn-on/Turn-off: Each regulator shall be turned on and turned off by separate turn-on/turn-off pulses. The pulses shall be standard command pulses.

3.3.5.3 Failure Turn-Off: Each regulator shall be capable of turning itself off or be capable of being turned off by command in the event of any internal failure.

3.4 Telemetry and Command Subsystem: The telemetry and command subsystem shall provide facilities to receive, process, and execute commands which will control spacecraft operation. It shall also provide facilities to encode digital and analog signals, which indicate quality of operation, and transmit these signals to the ground.

3.4.1 Reliability: The telemetry and command subsystem shall have a probability of operation within the performance requirements of

0.99 for a 1-year requirement

0.98 for a 3-year requirement

3.4.2 Telemetry and Command Antenna, 475045

3.4.2.1 Polarization: The polarization of the radiation shall be elliptical. For transmission, the axial ratio of the polarization along the spin axis shall not be greater than 1 db and, as a design objective (not a requirement), the ratio should be less than 3 db to a 30-degree angle with respect to the spin axis. For reception, the axial ratio of the polarization ellipse should be less than 3 db along the spin axis, as a design objective (not a requirement).

3.4.2.1.1 Radiation Pattern: As nearly isotropic a coverage as possible shall be provided. Radiation pattern measurements shall be made with linearly polarized source antennas.

TABLE 6-8. PACE DIGITAL CONTROL - 475161

Control Item	No. of Quadrants	Function	Description
INPUT FROM:			
475211	4	Central timer FF No. 1	Command pulse
475211	4	Central timer FF No. 2	Command pulse
475211	4	Backup command	Command pulse
475303	4	Selection logic	
475211	4	C1 Jet fire angle	Command pulse
475211	4	C2 Jet fire angle	Command pulse
475211	4	$\overline{C1}$ Jet fire angle	Command pulse
475211	4	$\overline{C2}$ Jet fire angle	Command pulse
475211	4	C3 Jet fire angle	Command pulse
475211	4	C4 Jet fire angle	Command pulse
475302	4	ψ_1	
475302	4	ψ_2	
475160	1	Power	+ 24 volts
475160	1	Power	- 24 volts
	4	Timing signal	
OUTPUTS TO:			
475221	4	ψ_2 angle	
475221	4	ψ_2 angle	
475221	4	ψ_2	
475221	4	Beam angle	
475221	4	Beam angle	
475221	4	Beam angle	
Axial jet No. 1		Jet command	
Axial jet No. 2		Jet command	
Radial jet No. 1		Jet command	
Radial jet No. 2		Jet command	
475152	4	Phase shift drivers $n = 0$	
475152	4	$n = 1$	
475152	4	$n = 2$	$20 \cos 2\pi$
475152	4	$n = 3$	$[\sin(2\pi ft + m\pi/8)]$
475152	4	$n = 4$	
475152	4	$n = 5$	
475152	4	$n = 6$	
475152	4	$n = 7$	
475152	4	$n = 0$	
			$20 \sin 2\pi$ $[\sin(2\pi ft + m\pi/8)]$

TABLE 6-8 (continued)

Control Item	No. of Quadrants	Function		Description
475152	4	Phase	n = 1	
475152	4	shift	n = 2	
475152	4	drivers	n = 3	
475152	4		n = 4	
475152	4		n = 5	
475152	4		n = 6	
475152	4		n = 7	
475221	4	PACE lock		

The standard command pulse shall be 60 msec long, have 5 msec rise time, and be 0.2 volt in amplitude.

3.4.2.1.2 Bandwidth: The center frequency shall be the transmission frequency. Signal strength of the receiving frequency shall not be down more than 3 db.

3.4.2.2 Diplexer

3.4.2.2.1 Quantity: Four diplexers shall be provided.

3.4.2.2.2 Frequency: Two channels shall be provided in each diplexer. One channel shall serve as an input to the receiver and the other channel shall serve as an output from the transmitter.

3.4.2.2.3 Isolation: The receiver channel shall offer at least 60 db rejection to the transmitter frequency. The transmitter channel shall offer at least 30 db rejection to the receiver frequency.

3.4.2.2.4 Insertion Loss: The insertion loss, with all diplexers, multiplexers and cables installed and properly terminated, shall not exceed 3 db between the input to any transmitter channel and the antenna terminals, or 3 db between the antenna terminal and the input terminals to the receiver.

3.4.2.2.5 Voltage Standing-Wave Ratio: The voltage standing-wave ratio at the receiver and transmitter terminals, with all necessary cabling, multiplexers, diplexers and antennas properly installed on the spacecraft, shall not exceed 1.5:1 at the transmitting frequency and 2.5:1 at the receiving frequency.

3.4.3 Command Group, 475050: The command group shall be composed of

1) Four Command Receiver, 475210

2) Four Command Filter-Decoders, 475211

3) Four Command Regulators, 475212

3.4.3.1 Command Receiver, 475210

3.4.3.1.1 Input Signal: The receiver shall be designed to receive amplitude modulated signals.

3.4.3.1.2 Frequency: The center of the frequency pass band shall be _______mc.

3.4.3.1.3 Noise Figure: The noise figure of the receiver shall be 10 db maximum referenced to the standard source temperature of 290°K.

3.4.3.1.4 Receiver Stability: The receiver shall remain within 0.003 percent of the selected frequency.

3.4.3.1.5 IF Bandwidth: The bandwidth at points 3 db down from maximum response shall be 60 kc ±15 kc.

3.4.3.1.6 Input and Output Impedance: The input and output impedance shall be 50 ohms.

3.4.3.1.7 Sensitivity: For amplitude modulated input signal levels of -95 dbm or greater the receiver output shall be sufficient to operate all command functions.

3.4.3.1.8 Input-Output: Command receiver inputs-outputs shall be as given in Table 6-9

3.4.3.2 Command Decoder, 475211

3.4.3.2.1 Operation: The command decoder shall process the audio signal so as to generate command signals to all required circuitry.

3.4.3.2.2 Command Pulse Duty Cycle: The command receivers and that portion of the decoder circuitry required to initiate full command turn-on shall be operating at all times that spacecraft power is on.

3.4.3.2.3 Redundancy: The command system shall be interconnected so that failure of one of the multiple systems does not compromise the ability of remaining systems to perform all command functions.

3.4.3.2.4 Real-Time Operation: The design of the command system shall be predicated on the necessity of a real-time RF link for the execute signal.

3.4.3.2.5 Inputs-Outputs: Command decoder inputs-outputs shall be as given in Table 6-10.

3.4.3.3 Command Regulator, 475212: The command regulator shall provide -24 volts ±1 percent to both the command receiver and command decoder. It shall also provide +24 volts ±1 percent to the command decoder.

TABLE 6-9 . COMMAND RECEIVER - 475210

Inputs and Outputs

Control Item	Number of Quadrants	Function	Description
INPUTS FROM:			
Whip antenna		Command receiver	148 mc
475212	1	Power supply	-24 volts
OUTPUTS TO:			
475221	4	AGC	
475211	1	0, 1, execute tones	Awaiting NASA frequency determination

TABLE 6-10. COMMAND DECODER - 475211

Control Item	Number of Quadrants	Function	Description
INPUTS FROM:			
475210	1	0, 1, execute tones	Awaiting NASA frequency determination
475212	1	Power supply	±24 volts regulated
475212	1	Power supply	-24 volts unregulated
OUTPUTS TO:			
475160	4*	Turn on series regulator	Command pulse
475160	4*	Turn off series regulator	Command pulse

TABLE 6-10 (continued)

Control Item	Number of Quadrants	Function	Description
475025	4*	Turn on multiple-access series regulator	Command pulse
475025	4*	Turn off multiple-access series regulator	Command pulse
475025	4*	Turn on frequency translation series regulator	Command pulse
475025	4*	Turn off frequency translation series regulator	Command pulse
475175	4*	Turn on series regulator for TWT No. 1	Command pulse
475175	4*	Turn off series regulator for TWT No. 1	Command pulse
475175	4*	Turn on series regulator for TWT No. 2	Command pulse
475175	4*	Turn off series regulator for TWT No. 2	Command pulse
475160	4	Central timer select FF No. 1	Command pulse
OUTPUTS FROM:			
475160	4	Central time select FF No. 2	Command pulse
	4	Jet fire backup command	Command pulse

TABLE 6-10 (continued)

Control Item	Number of Quadrants	Function	Description
475160	4	C1	Command pulse
	4	$\overline{C1}$ Jet fire angle	Command pulse
	4	C2	Command pulse
	4	$\overline{C2}$	Command pulse
	4	C3	Command pulse
	4	C4	Command pulse
475174	4*	High voltage start TWT No. 1	Command pulse
	4*	High voltage start TWT No. 2	Command pulse
475160	4	Command antenna beam angle	Command pulse
475221	4	Execute command	Command pulse
475221	4	Encoded command register	
475221	4	Encoded command register	
475221	4	Encoded command register	
475221	4*	Turn on	Command pulse
	4*	Turn off	Command pulse

*One distinct and different signal shall be sent to each of the quadrants. The standard command pulse shall be 60 msec long, have a 5 msec rise time, and be 0.2 volt in amplitude.

3.4.4 Telemetry Group, 475055: The telemetry group shall consist of:

1) Four Telemetry Transmitters, 475220

2) Four Telemetry Encoders, 475221

3) Four Telemetry Regulators, 475222

3.4.4.1 Telemetry Transmitter, 475200

3.4.4.1.1 Frequency: Two of the transmitters shall be designed to operate at a frequency of _______ mc and the other two transmitters should be designed to operate at a frequency of _______ mc.

3.4.4.1.2 Frequency Stability: The transmitter frequency shall remain within 0.003 percent under normal service conditions.

3.4.4.1.3 Power Output: The transmitter output shall be at least 1 watt into a 50-ohm load.

3.4.4.1.4 Modulation: The modulator portion of the transmitter shall angle-modulate the RF signal generated in the transmitter.

3.4.4.1.5 Modulation Sensitivity: The modulation sensitivity shall provide 1.5 radians phase deviation for normal input from the encoder.

3.4.4.1.6 Input and Output Impedance: Input and output impedance shall be 50 ohms.

3.4.4.1.7 Inputs - Outputs: The telemetry transmitter inputs-outputs shall be as given in Table 6-11.

TABLE 6-11. TELEMETRY TRANSMITTER - 475220

INPUTS FROM:	
Same as outputs from 475221 and -24 volt power supply from 475222	
OUTPUTS TO:	
475201	Same as inputs except for -24 volts

3.4.4.2 Telemetry Encoder 475221: The telemetry encoder shall be capable of commutating the input signals (Table 6-11) and providing a suitable modulation signal to the telemetry transmitter.

3.4.4.2.1 Analog Signals: The analog signals shall be ________ ______________.

3.4.4.2.2 Digital Signal: The digital signals shall be __________ volts ____________ percent, have a ________ rise time, and be ________ long.

3.4.4.2.3 Input and Output: The telemetry encoder inputs-outputs shall be as given in Table 6-12 .

3.4.4.3 Telemetry Regulator 475200: The telemetry regulator shall provide -24 volts ±1 percent. It shall be a series-type regulator.

3.4.4.3.1 Turn-on/Turn-off: The telemetry regulator shall be capable of being turned on and turned off by a standard command pulse from any of the four command decoders.

TABLE 6-12. TELEMETRY ENCODER - 475221
Inputs and Outputs

Control Item	Number of Quadrants	Function	Description
INPUTS FROM:			
475065		Tranverse acceleration	Analog
475065		Tranverse acceleration	Analog
475065		Longitudinal acceleration	Analog
475061	4	Jet fire	Digital
475211	4	Execute	Digital
Battery No. 1		Unregulated bus voltage No. 1	Analog
Battery No. 2		Unregulated bus voltage No. 2	Analog
475252		Solar panel temperature	Analog

TABLE 6-12 (continued)

Control Item	Number of Quadrants	Function	Description
475161	4	PACE lock	Digital
475161	4	Antenna beam angle	Digital
475161	4	Antenna beam angle	Digital
475161	4	Antenna beam angle	Digital
475211	4	Command verification	Digital
475211	4	Command verification	Digital
475211	4	Command verification	Digital
475302	4	ψ_2 angle	Digital
475302	4	ψ_2 angle	Digital
475302	4	ψ_2 angle	Digital
	4	Propellant tank pressure No. 1	Analog
	4	Propellant tank pressure No. 2	Analog
	4	Propellant tank pressure No. 3	Analog
	4	Propellant tank pressure No. 4	Analog
475172		Transmitter power No. 1	Analog
475172		Transmitter power No. 2	Analog
475172		Transmitter power No. 3	Analog
475172		Transmitter power No. 4	Analog

TABLE 6-12(continued)

Control Item	Number of Quadrants	Function	Description
475025		Receiver signal strength No. 1	Analog
475025		Receiver signal strength No. 2	Analog
475025		Receiver signal strength No. 3	Analog
475025		Receiver signal strength No. 4	Analog
475040	4	Receiver - TWT selection	Digital
475220	1	Telemetry radiated power	Analog
475065		Temperatures	Analog
475065		Temperatures	Analog
475065		Radiation experiment	Analog
475065		Radiation experiment	Analog
475065		Radiation experiment	Analog
475065		Radiation experiment	Analog
475065		Radiation experiment	Analog
47522	1	Power supply	-24 volts
475302	4	ψ	Analog
475302	4	ψ_2	Analog

TABLE 6-12 (continued)

Control Item	Number of Quadrants	Function	Description
OUTPUTS TO:			
475220	Same as inputs excluding -24 volts (power supply) and including spacecraft identification, T/M set identification, and calibration reference.		

3.4.5 General Subsystem Requirements

3.4.5.1 Countdown Testing: The beacon tracking, telemetry, and command subsystem shall operate with the nose fairing in place to the extent necessary to control and measure proper performance of the spacecraft.

3.4.5.2 Operating Duty Cycle: The telemetry and command subsystem shall be designed to have the capability of continuous operation.

3.4.5.3 Telemetry and Test Signal Provisions: Test plugs shall be provided on the units of the telemetry and command subsystem as required to allow appropriate signal voltages to be monitored for telemetry and test purposes.

3.5 Power Supply Subsystem, 475060: The electrical power subsystem shall provide the spacecraft electronics with operating power. The electrical power will be provided by solar panels. A battery shall be provided to supply power during eclipses. There shall be a battery regulator to control charge current. The subsystem shall consist of

1) 96 battery cells

2) 16 solar panels

3) 4 battery regulators

3.5.1 Reliability: The power subsystem shall have a probability of operation within the performance requirements of

0.998 for a 1-year requirement

0.994 for a 3-year requirement

3.5.2 Battery Regulator, 475251

3.5.2.1 Quantity: There shall be four battery regulators.

3.5.2.2 Type: The regulators shall be of a boost type.

3.5.2.3 Charge Voltage: There shall be at least 36 volts for charging of 24 cells.

3.5.2.4 Charge Current: The maximum charge rate shall be 300 ma. The maximum trickle current shall be 20 ma.

3.5.2.4.1 Availability of Current: The maximum current available shall be not more than 1 ampere for charging of 96 cells.

3.5.3 Solar Array

3.5.3.1 Quantity: There shall be sixteen solar panels. They shall consist of fourteen Solar Panels - 475252 and two Solar Panels, Special 475253.

3.5.3.2 Common Requirements

3.5.3.2.1 Solar Array Output: The solar array voltage output shall be 27.0 volts and 5.14 amperes. The above power requirement shall be at a solar intensity of 140 mw/cm,^{2} a temperature of 77°F and a sun incidence angle of 90 degrees ±25 degrees.

3.5.3.3 Peculiar Requirements

3.5.3.3.1 Solar Panel, Special, 475253: The special solar panels shall be similar to the Solar Panel 457252, having a different cell layout to accommodate protrusion of the radial jet.

3.5.4 Battery

3.5.4.1 Quantity: There shall be 96 nickel-cadmium battery cells.

3.5.4.2 Discharge-Charge Efficiency: The discharge-charge efficiency is defined as the ratio of the ampere-hours removed from a fully charged battery during discharge to the ampere-hours required to restore it to its originally fully charged condition. For the initial system design, the discharge-charge efficiency shall be greater than 36 percent (including change regulations).

3.5.4.3 Power Capacity: The capacity shall be:

26 volts

6.0 amp-hr at 75° F at 1.2 amperes

4.8 amp-hr at 100° F at 1.2 amperes

4.8 amp-hr at 30° F at 1.2 amperes

3.5.4.4 Maximum Charge Rate: The maximum charge rate shall be 5 amperes. The maximum overcharge rate shall be 0.6 ampere continuous.

3.5.4.5 Maximum Charge Voltage: The maximum charge voltage shall be 1.48 volts.

3.5.4.6 Cycle Life: The minimum cycle life shall be 10,000 at 25 percent depth.

3.5.5 Power Distribution

3.5.5.1 The power distribution shall be as given in Table 6-13.

TABLE 6-13. POWER DISTRIBUTION

Subsystem	Number of Units per Quadrant	Total Units Operating	Milli-amperes per Units	Milliamperes per Bus Load
Telemetry	1	1	245	245
Command Receiver	1	4	27	108
Encoder	1	1	27	27
Antenna Electronics	1	1	650	650
Communication Receivers	2	4	75	300
Traveling-Wave Tubes (4 watt)	2	4	693	2772
Battery Charging	4	4		620
Total Bus Load				4722

3.5.5.2 Unregulated Bus Characteristics

3.5.5.2.1 Voltage: The unregulated bus voltage shall be within the limits of -27 and -36 volts dc during normal operating conditions in orbit with all equipment functioning properly. Necessary power regulation or conversion shall be provided as part of each subsystem.

3.5.5.2.2 Transients: During transfer of the spacecraft equipment loads from any mode of operation to another the voltage at the equipment terminals shall remain within the range of -24 to -37 volts dc and shall recover and remain within the steady-state limits in less than 0.5 second.

3.5.5.2.3 Ripple: The peak-to-peak ripple voltage output of the solar panels, measured on the unregulated bus, shall not exceed 0.5 volt. The ripple frequency shall be less than 1000 cps.

3.5.6 Discharge Control: Discharge control shall be provided by dropout of the loads under reduced voltage input. The dropout voltage shall increase when loads over the rated value are placed on a subsystem regulator by defective circuitry. The removal of loads by regulator dropout shall not impair later normal use of the subsystem when the voltage is restored to normal values.

3.6 Structure Subsystem

3.6.1 Structure: The spacecraft structure shall provide the basic support for the other subsystems of the spacecraft and for the attachment to the spacecraft support structure of the boost vehicle.

3.6.1.1 Accessibility: Accessibility and the capability of quick removal of components shall be considered in the design for mounting subsystems. The covers of the spacecraft shall be removable to permit maximum accessibility to the internal subsystems for replacement, repair, and checkout with minimum weight expenditure.

3.6.1.2 Appendages: Four tracking, telemetering, and command antennas shall be mounted on the forward (direction of launch or +Z) end of the spacecraft. The communications antenna shall be mounted on the aft (or -Z) end of the spacecraft on the spin axis.

3.6.1.3 Use of Shock and Vibration Isolators: The use of shock and vibration isolators shall require approval of the GSFC Project Manager.

3.6.2 Thermal Control Requirements: Thermal control of the spacecraft shall be accomplished by passive and/or active temperature design of the structure. Temperatures shall be controlled on each spacecraft part, unit, or subsystem within a range compatible with its function and its reliability requirements.

3.6.3 Reliability: The structure subsystem shall have a probability of operation, within the performance requirements of

0.997 for a 1-year requirement

0.991 for a 3-year requirement

3.6.4 Nutation Damper, 475301: The spacecraft shall utilize two nutation dampers to dissipate the spacecraft nutation energy as heat. The design shall limit the maximum nutation angle to 1° when the spacecraft is precessed 135°. The time constant of this device shall be less than 1 hour.

3.6.5 Spin Rate Control: Provision shall be made for controlling the spin rate to within 100 ± 50 rpm.

3.6.6 Sun Sensor Assembly, 475302

3.6.6.1 Four sun sensor assemblies shall be provided. The sun sensors shall provide a means of determining the angle between the spin axis and the sun line as well as synchronization data on the spacecraft spin rate.

3.6.6.2 Coverage: The sun sensors shall provide an output in each of the four quadrants. The roll-angle sensors shall provide maximum outputs at 0, 90, 180, and 270 degrees when the rays of the sun are normal to the spacecraft spin axis. The attitude roll-angle sun sensors shall be paired with the sun sensors which are adjacent to one of the hot gas axial control jets. The major axes of the attitude sun sensor shall be inclined at 35 degrees to the spin axis.

3.6.6.3 Beam Width: The 3-db beam width of the sun sensors shall be 0.8 degree by not less than 150 degrees.

3.6.6.4 Output Voltage: The output voltage shall be at least 0.18 degree by 150 degrees minimum.

3.6.6.5 Alignment: The beam planes of the roll angle sun sensors shall be parallel to the spin axis of the spacecraft to within 0.5 degree. The beam plane of each attitude sensor (ψ_2) shall be set to a value of 35 degrees ± 0.5 degree with respect to its reference sensor (ψ). The combination of sensors shall be capable of measuring the angle between the spin axis and the sun line within 1 degree when the angle is within 90 ± 25 degrees.

3.6.7 Pyrotechnic Switch Assembly: Two pyrotechnic switch assemblies are used to ensure that an open circuit exists between the unregulated bus and the apogee motor squibs after the squibs have been fired. The requirements for this assembly are contained in the Procurement Specification.

3.6.8 Separation Switch: Four separation switches shall be provided to define by telemetry separation of the spacecraft from the Agena and to provide power to the central timer. The requirements for the item are contained in the Procurement Specification.

3.7 Wire Harness Subsystem, 475300

3.7.1 Wire

3.7.1.1 Voltage Drop: The electrical power and signal distribution system shall be so designed that at no time shall any terminal voltage fall below the rated value due to excessive voltage drop in response to transmission of rated currents.

3.7.1.2 Mechanical Strength: No wires smaller than size AWG-26 or equivalent etched circuit lines shall be used in the electrical power and signal distribution system.

3.7.1.3 Installation of Wiring: The installation of wiring shall be in general accordance with the applicable requirements of specification MIL-W-8160.

3.7.2 Harness Construction: The harness shall be constructed to minimize noise effects.

3.7.3 Twisting: Twisting shall be used whenever necessary to eliminate noise effects.

3.7.4 Shielded Wire: As a general rule, there shall be no electrical connections between the shield of any shielded wire and any electrical circuitry. As a general rule, the shield of a shielded wire shall be grounded only at one end.

3.7.5 Grounding

3.7.5.1 Common Ground: System return leads requiring grounding shall be terminated as close as possible to the positive battery ground terminal.

3.7.5.2 Unit Case Grounds: A component case may be considered a shield. One connector pin on each unit may be electrically connected to the unit case internally. The case may be grounded by mechanically contacting a system ground plane or by wiring to the case ground connector pin.

3.7.6 Soldering: All soldering operations that are to be made on Syncom II must be done by personnel that have been certified as passing the requirements set forth in NASA Document MSFC-PROC-158B, dated 15 February 1963. Subject, "Procedure for Soldering of Electrical Connectors." (High Reliability)

The following numbers indicate paragraph numbers of the referenced specification to which exception shall be taken.

VI A 2c	If tubing cannot be used at abrasion points, wrapping may be substituted. If tubing cannot be fitted over a terminal, it may be dispensed with if the terminal has been correctly soldered, and sufficient clearance exists. MIL-I-22129 is waived.
VI A 1d	Preparation of new terminals should not be necessary. Terminals that are being reworked may be cleaned as necessary, care being taken not to damage the terminal or surrounding area.
VI A 1g	Silver-plated wire presently utilized on Syncom II is insulated with fluorinated ethylene propylen (FEP), and is acceptable.
VI A 2a	Any thermal stripper, such as American Missile Products MOD. WS-17B, Ideal Model No. 45-130 or 45-141, that, in the opinion of the Quality Control inspector, does a satisfactory job, is authorized.
VI A 2d	Heat sink tools will continue to be used. However, the circumferential edges of the tool will be rounded and burnished to reduce the possibility of wire abrasion.
VI A 2h	Resistance soldering will not be used until it has been thoroughly tested and established as a satisfactory technique.
VI A 3b	The environmental conditions described cannot at present be met. However, all good housekeeping procedures will be followed, as described in VI A 3a. To reduce the possibility of accidental damage from loose tools, soldering technicians will use tool trays. Harness boards will be cleaned regularly to avoid inclusions of debris in the harness assemblies.
VI A 3e(2)	The exposed wire between pot and insulation shall be limited to 0.1 inch. If by chance this tolerance is exceeded, NASA will consider and may approve the condition on an individual basis.

VI B 1d — The solder should follow the cup contour as closely as is reasonable under the particular circumstance, and should have a slightly concave appearance. However, a convex configuration is acceptable provided the solder does not protrude beyond the outer diameter of the pot. It is recognized that the soldering process may leave a small amount of solder adhering to the outer surface of the pot, but excessive amounts, in the opinion of the Quality Control inspector, shall not be allowed. Inspections will be made with five to ten power magnifiers. In cases of controversy a higher power magnification may be used.

VI B 2a — Where design or layout necessitates the connecting of more wires than the terminal was designed for, the extra wires may be wrapped around the terminal. All caution will be used to ensure proper clearance, insulation, and correct soldering. A sufficient excess length of wire may be used on terminal posts to permit one wiring change. These connections shall be heavily coated with protective paint to provide support as well as insulation.

VI B 1f — Wicking should not extend beyond 0.25 inch from the solder pot and connections which exceed this limit will be rejected if the insulation is bulged.

VI B 3 — Where necessary, in the opinion of the Quality Control inspector, the wrap may be increased up to 270°.

3.7.7 Filtering: RF filters shall be provided on power line inputs to RF circuitry.

3.8 Apogee Motor Subsystem: This subsystem shall consist of a solid propellant rocket engine to be used for injection of the Syncom II spacecraft into a nominally circular equatorial orbit from the apogee of a transfer orbit. This subsystem is GFE. The requirements for this subsystem are defined in Procurement Specification for Syncom II Apogee Rocket Motor; Buyer: National Aeronautics and Space Administration, Contractor: Jet Propulsion Laboratory.

3.9 Reaction Control Subsystem: The reaction control subsystem shall provide two redundant sources of thrust to correct spacecraft longitude, to provide the spacecraft with the capability of being synchronous in the equatorial plane, maintain spacecraft spin rate and orient the spacecraft so that the antennas will illuminate the earth continuously.

The unit shall consist of storage tanks for fuel and oxidizer, injector solenoid valves, fuel and oxidizer lines, and thrust chambers aligned axially and radially. The requirements for this subsystem are defined in Hughes Specification X-254044, Procurement Specification for Syncom II Bipropellant Reaction Control System.

Preliminary Design of a Simplied Transponder

A transponder design based on the new channel allotments has been formulated and is shown in Figure 6-5. In addition to the simplifications made possible by the present frequencies and the modifications required to incorporate a coherent beacon, some changes have been dictated by the desire to improve system performance. A ferrite switch has been added to the input, completely separating the two receivers rather than depending on a back-biased crystal to control signal flow. Several attenuators have been added to both the frequency translation and multiple access transponders for better control of signal level and increased isolation between units. An additional bandpass filter has been added at the output of each transponder so that the traveling-wave tube will amplify only signals in the desired frequency range. The packaging of some IF amplifiers has also changed, resulting in additional units. The limiter amplifier is being modified to accept a beacon input in the same manner as on Syncom I. The resulting beacon signal will thus be coherent.

Figure 6-5 indicates the expected signal levels at various points in the system. In addition, losses are shown wherever they occur. Frequencies are also given for pertinent units to aid in understanding the system.

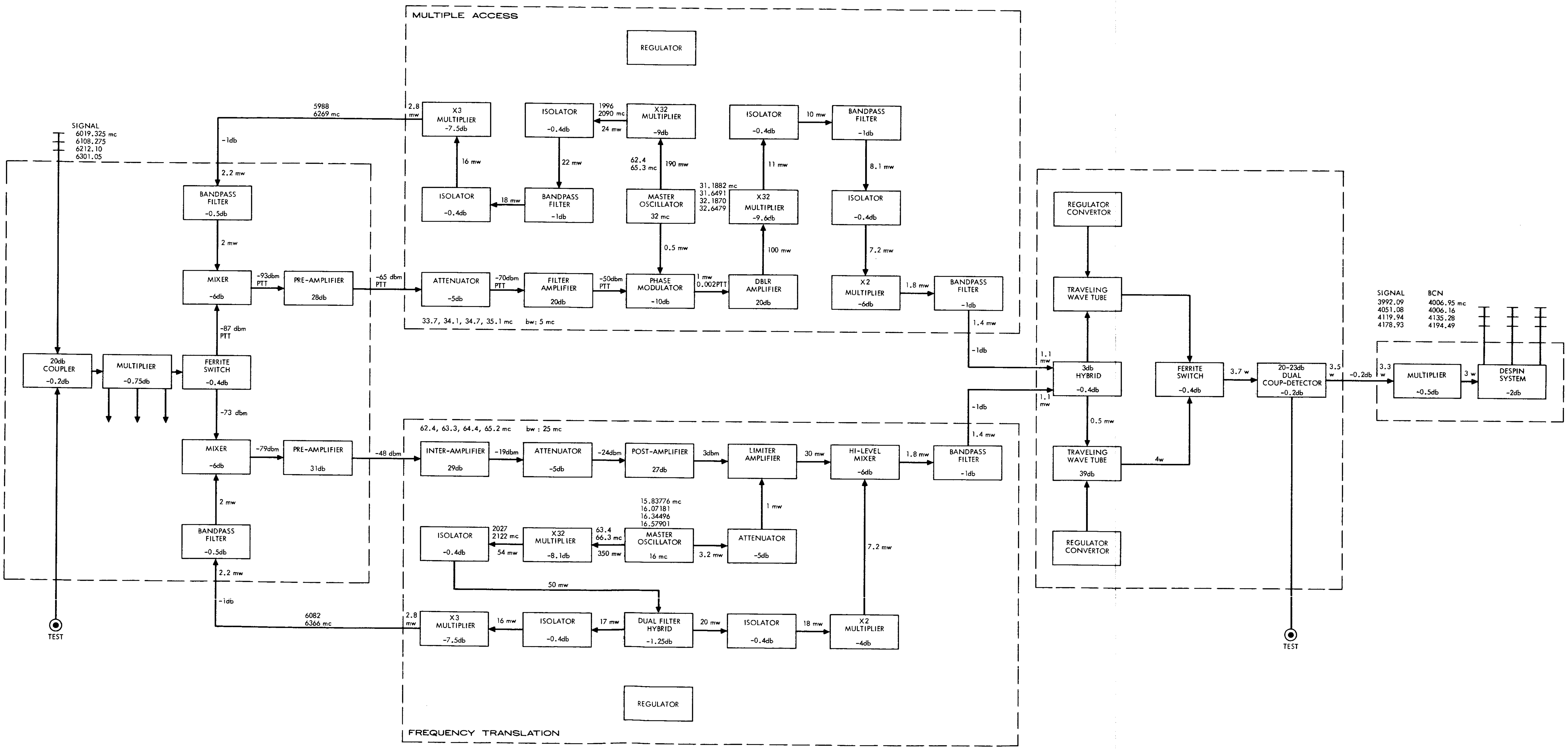


Figure 6-5. Block Diagram for Syncom II Transponder System

Dual-Mode Transponder

Engineering data testing of transponder components, as a function of various environmental conditions, has been accomplished. Data concerning the components was gathered while varying the temperature, signal level, and power supply voltages. All components tested were not final configuration and in several instances were laboratory breadboards which were not optimized for such parameters as noise figure and temperature compensation.

Circuits Common to Frequency Translation and Multiple-Access Transponders

Input Mixer. The noise figure over the passband was measured at 25° C. The noise figure at 6.2 Gc, as a function of temperature, was measured. The results are shown in Figure 6-6.

Input Filter. The insertion loss and frequency range were checked at 0, 25, and 50° C. The results are shown in Table 6-14.

Local Oscillator Filter. The insertion loss and frequency range were checked at 0, 25, and 50° C. The results are shown in Table 6-15.

X2 Multiplier. The bandwidth and output power versus input power were measured at ambient temperature and are shown in Tables 6-16 and 6-17. Output power with constant input was measured at various temperatures and the results are shown in Table 6-18 . The VSWR was 1.3:1 and no spurious oscillations were observed over an input power deviation of ± 3 db.

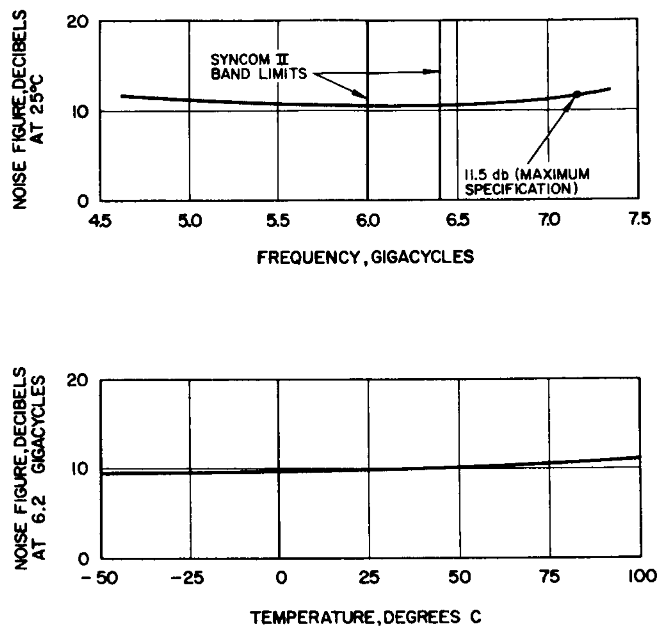
X3 Multiplier. Input power versus output power as a function of temperature and bias voltage was measured. The results are shown in Table 6-19.

Circuits for FM Frequency Translation Transponder

25-mc IF Preamplifier and Postamplifier. The gain, linearity, and bandpass were checked at three bias levels at 0, 22, and 50° C. The results are shown in Table 6-20 and in Figure 6-7.

High-Level Mixer. The power output with constant input power was checked at various temperatures. The results are shown in Table 6-21.

Master Oscillator. Two master oscillators and X32 combinations were used to obtain a 0 beat frequency so that the short-term stability of the oscillator could be observed. The results are shown in Figure 6-8a. The long-term or free-running drift of the combination is shown in Figure 6-8b.



NOISE FREQUENCY MEASURED WITH "NOISY" 34mc IF PRE-AMPLIFIER (3.8 db N.F.) AND 1.2 db ADDITIONAL RF CABLE LOSS

Figure 6-6. Input Mixer Data

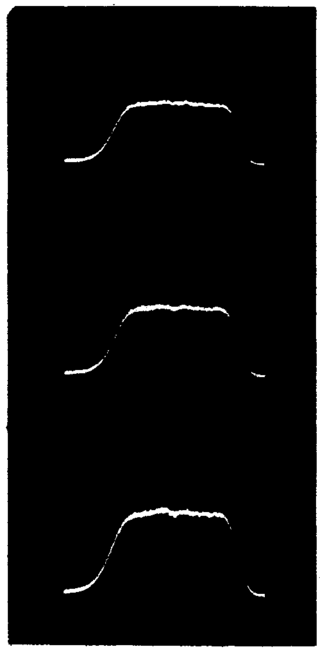
Engineering prototype stripline modulator

1) 1db POINT

2) 78 mc MARKER PIP

3) 48mc MARKER PIP

a) BANDWIDTH SPECTRUM



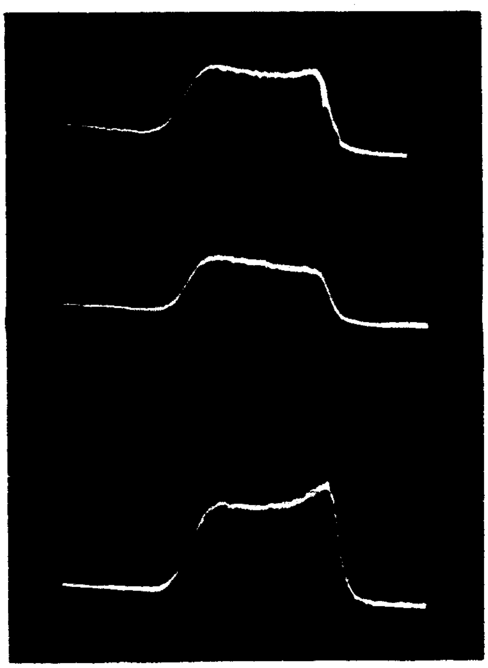
1) -22v COLLECTOR VOLTAGE

2) -24v COLLECTOR

3) -26v COLLECTOR

b) OUTPUT AMPLIFIER

a) Bandwidth spectrum



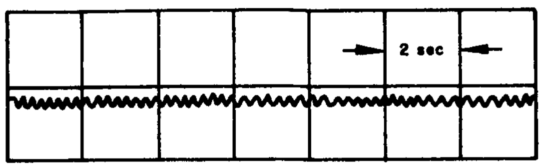
1) 22° C

2) 50° C

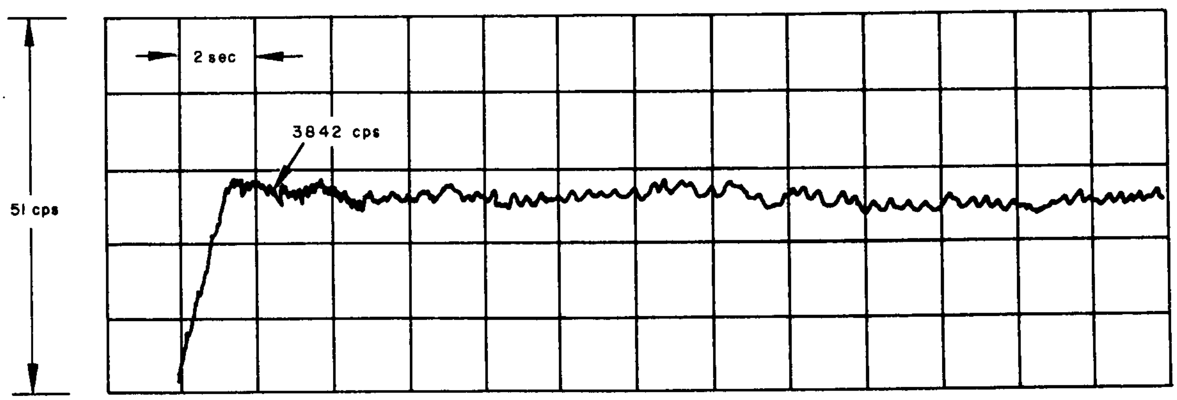
3) 0

b) Output amplifier

Figure 6-7. IF Preamplifier and Postamplifier Waveforms



a) Short-term stability



b) Long-term stability

Figure 6-8. Master Oscillator X32
Multiplier Bent Frequency

Dual-Filter Hybrid. The output-to-output ratio, total insertion loss, VSWR, isolation between outputs, and hybrid directivity were measured at 23° C. The insertion loss and rejection input to output were measured at 23, 0, and 50° C. The results are shown in Table 6-22.

Dual Single-Sideband Filter Diplexer. The insertion loss and rejection over the bandpass was measured at -25, 0, 23, 50, and 75° C. The results are shown in Table 6-23.

54-mc Wide-Band Limiter. Monitor output voltage and output power as a function of input power were measured at -25, 0, 25, 55, and 75° C. The results are shown in Tables 6-24 through 6-28 and Figure 6-9.

Multiple-Access Transponder

Single-Sideband Filter (2085 mc). The insertion loss, VSWR, and rejection over the band were measured at 0, 23, and 50° C. The results are shown in Table 6-29.

Filter (2119 mc). The insertion loss, VSWR, and rejection over the band were measured at 0, 23, and 50° C. The results are shown in Table 6-30.

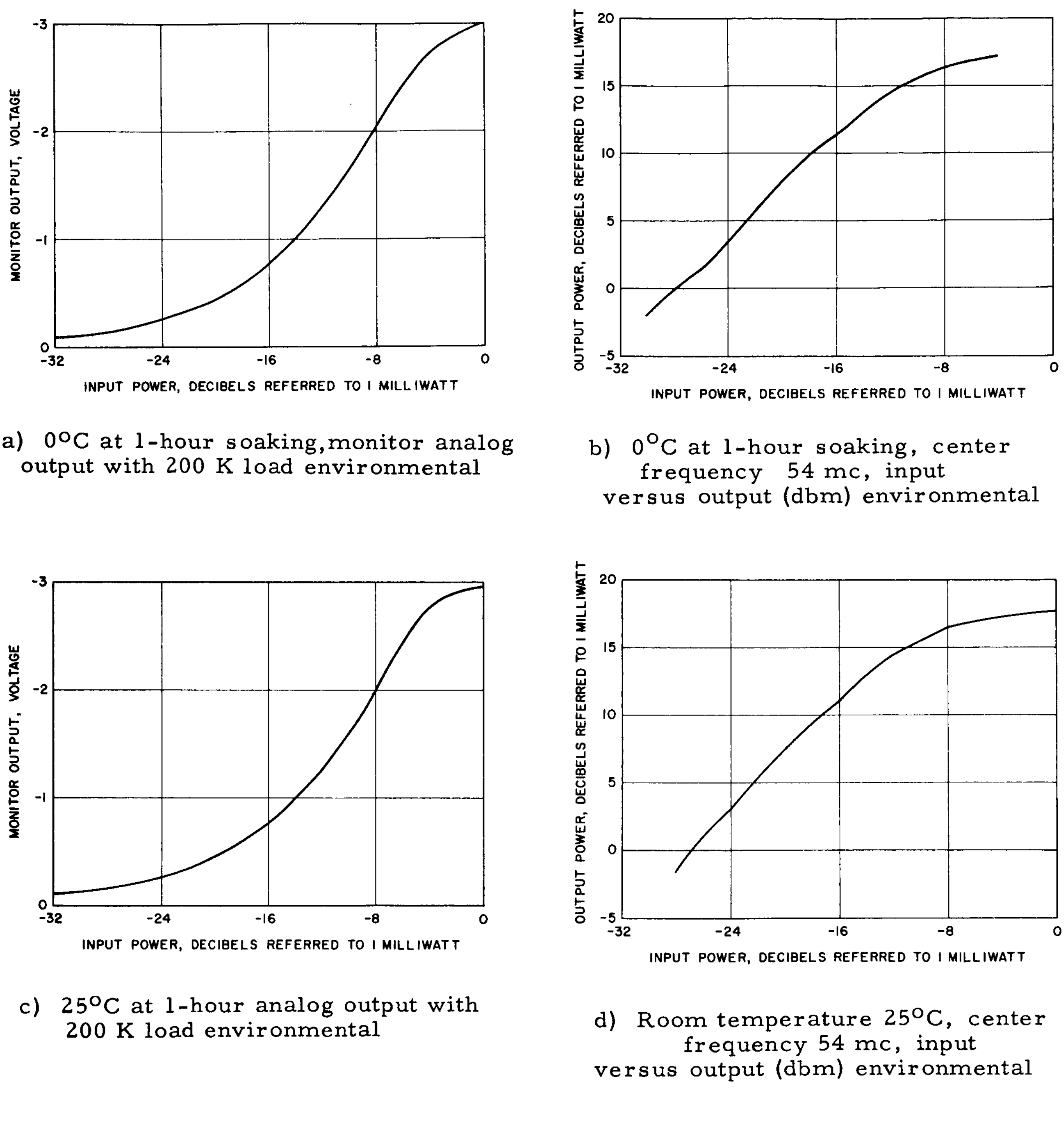
Master Oscillator. The frequency stability and power output were measured as a function of varying temperature. The results are shown in Figure 6-10.

Phase Modulator. The power output and modulation index were measured as a function of temperature. The results are shown in Figure 6-11.

Special Tests

The power output of the master oscillator, phase modulator, amplifier—X2 multiplier, X32 multiplier chain as a function of power supply voltage was measured at 0, 24, and 50° C. The results are shown in Figure 6-12.

The bandpass characteristics of the X32 multiplier and phase modulator as a function of supply voltage was measured at 0, 24, and 50° C. The results are shown in Tables 6-31 and 6-32 and in Figure 6-13.



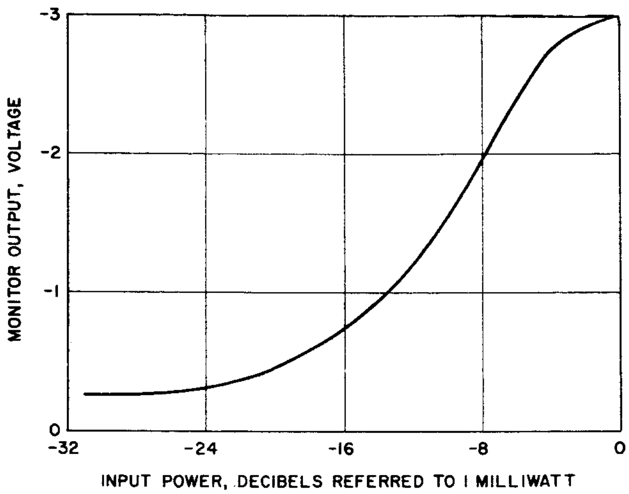
a) 0°C at 1-hour soaking, monitor analog output with 200 K load environmental

b) 0°C at 1-hour soaking, center frequency 54 mc, input versus output (dbm) environmental

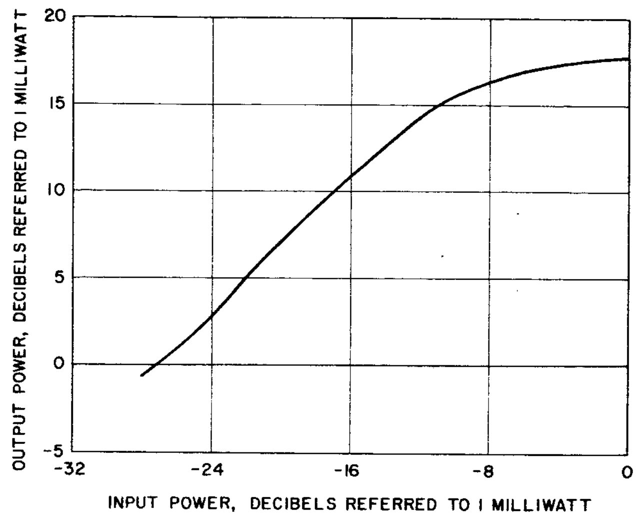
c) 25°C at 1-hour analog output with 200 K load environmental

d) Room temperature 25°C, center frequency 54 mc, input versus output (dbm) environmental

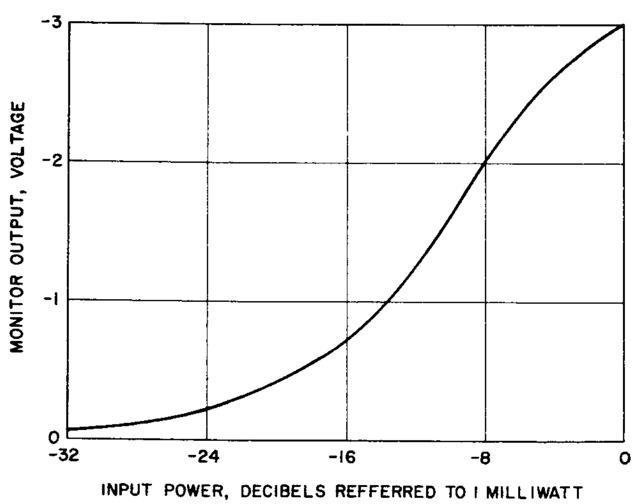
Figure 6-9. 54-mc Wide-Band Limiter



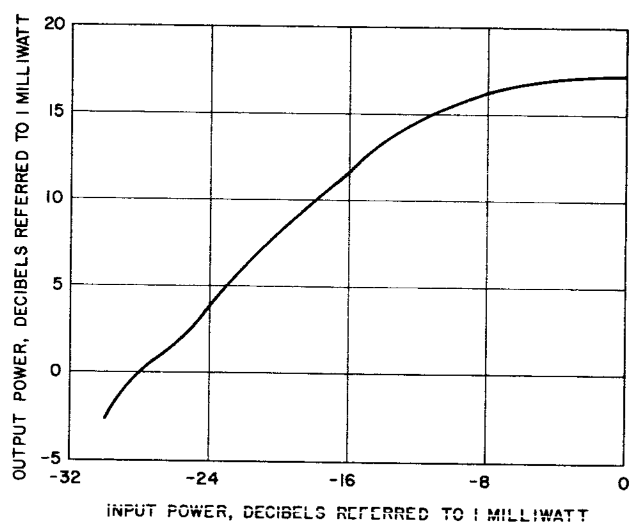
e) 55°C at 1-hour soaking, monitor analog output with 200 K load environmental



f) 55°C at 1-hour soaking, center frequency 54 mc, input versus output (dbm) environmental

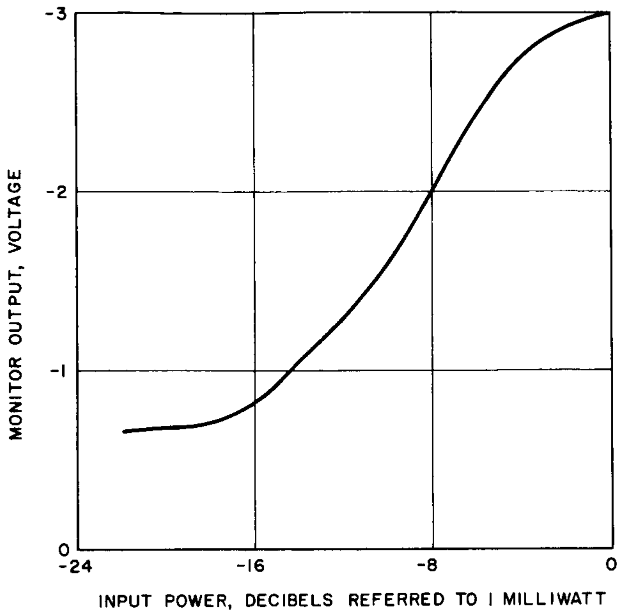


g) -25°C at 1-hour soaking, monitor analog output with 200 K load environmental

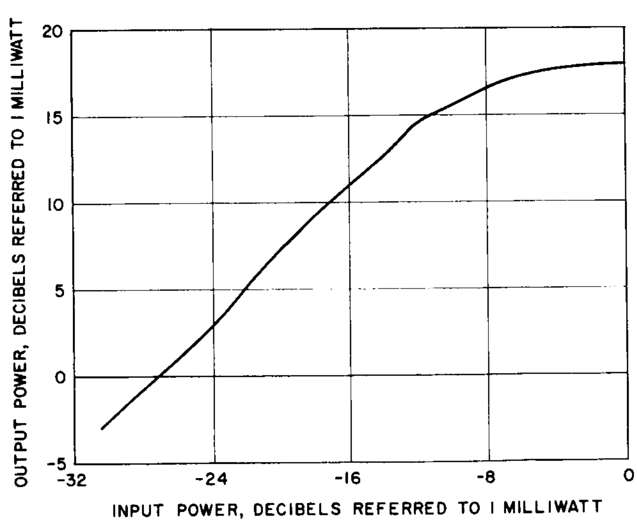


h) -25°C at 1-hour soaking, center frequency 54 mc, input versus output (dbm) environmental

Figure 6-9 (continued). 54-mc Wide-Band Limiter



i) 75°C at 1-hour soaking, monitor analog output with 200 K load environmental



j) 75°C at 1-hour soaking, center frequency 54 mc, input versus output (dbm) environmental

Figure 6-9 (continued). 54-mc Wide-Band Limiter

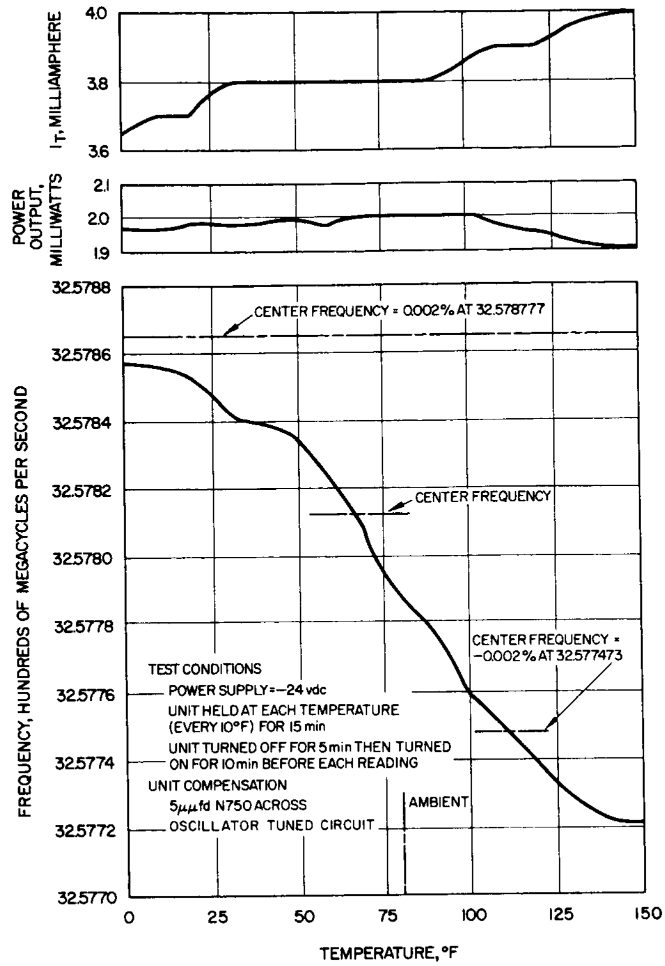


Figure 6-10. Temperature Test: Breadboard Transmitter Master Oscillator

Power supply voltage: -24 volts dc

TEST CONDITIONS

1) Power supply = -24 v
2) Oscillator input = 2 mw
3) IF input = -30 dbm
4) At each temperature, power disconnected and connected after 5 minutes of inoperation

REMARKS

1) At temperature of 0 to 40° F, carrier sideband greater by approximately 30 db when power reconnected. Application of excessive IF input returned carrier to sideband ratio to those indicated. No appreciable effect on output spectrum. Trouble attributed to loose trimmer capacitor making bad electrical connections to chassis.

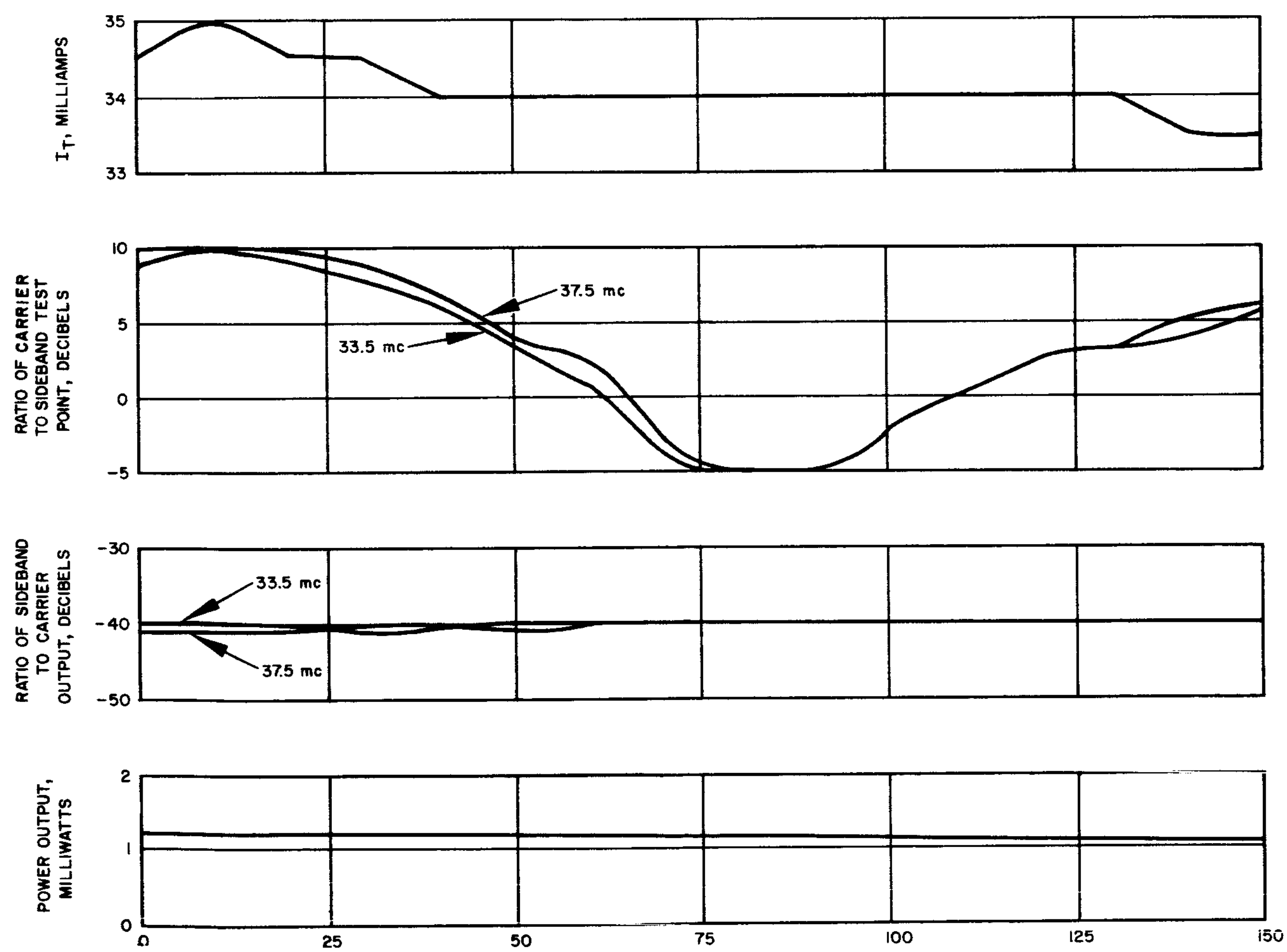


Figure 6-11. Temperature Test: Breadboard Phase Modulator

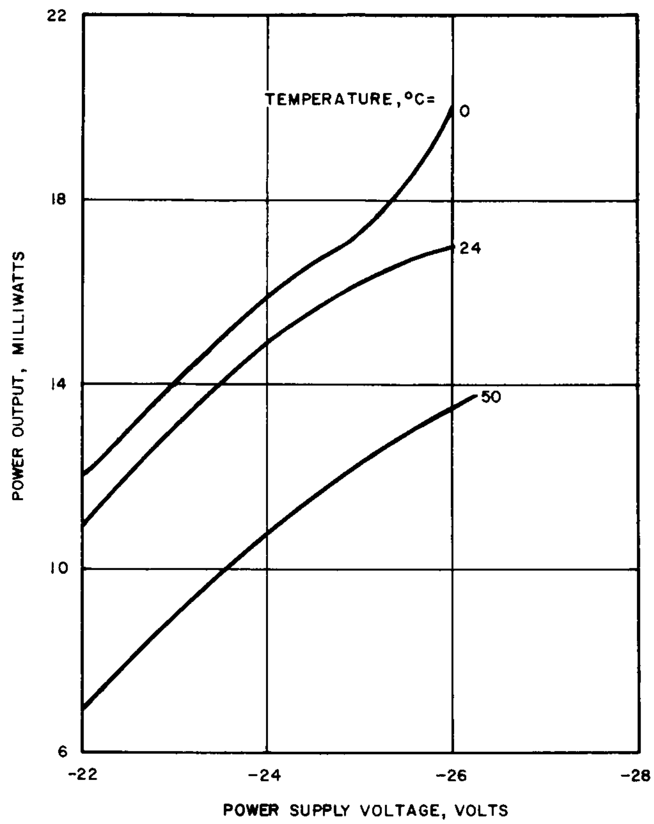
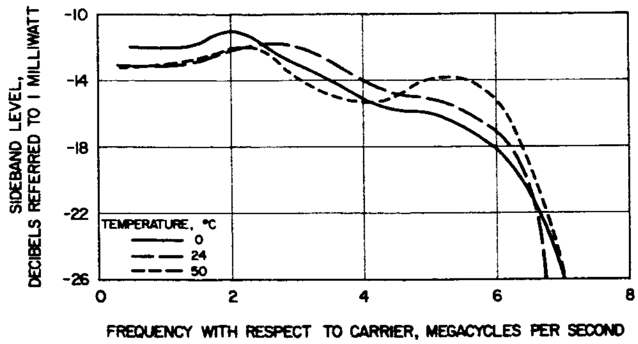
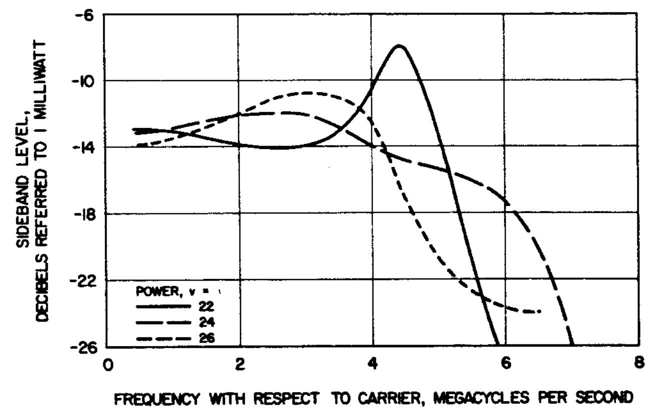


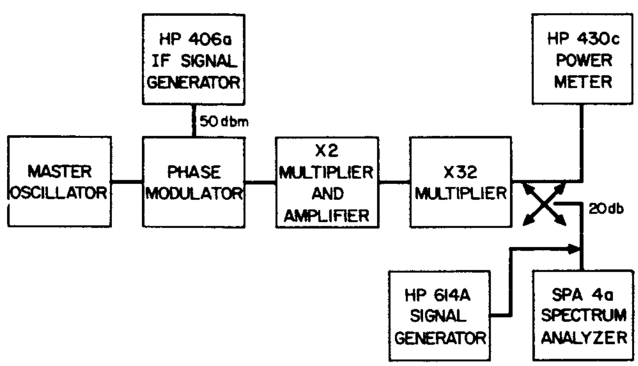
Figure 6-12. Power Output of Master Oscillator, Ø Modulator, Amplifier-X2 Multiplier Chain as Function of Power Supply Voltage



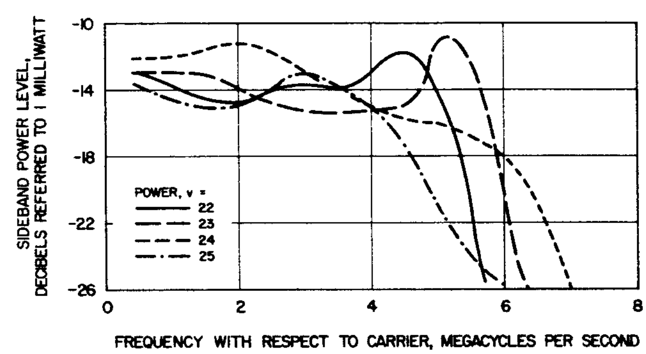
a) Ø modulator combination at 24 volts



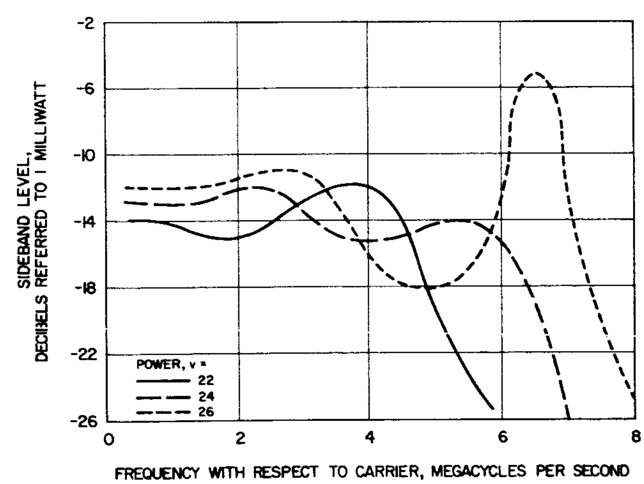
b) Ø modulator combination at 24°C



c) Block diagram of test setup



d) Ø modulator combination at 0°C



e) Ø modulator combination at 50°C

Figure 6-13. Bandpass Characteristic of X32 Multiplier

TABLE 6-14. INPUT FILTER TEST RESULTS

Temperature, °C	Insertion Loss, db	3-db Points, mc
0	1.3	6376, 6406
25	1.3	6374, 6404
50	1.3	6372, 6402

TABLE 6-15. LOCAL OSCILLATOR FILTER TEST RESULTS

Temperature, °C	Insertion Loss, db	3-db Points, mc
0	0.8	6335, 6344
25	0.8	6332, 6340
50	0.8	6330, 6338

TABLE 6-16. X2 MULTIPLIER BANDWIDTH

Tuned at 2050 mc, 15 mw Input, 5.6 mw Output

Frequency, mc	Output Power, mw	Loss, Relative to 5.6 mw, db
2030	4.4	1.0
2035	5.0	0.5
2040	5.4	0.2
2045	5.6	0.0
2050	5.6	0.0
2055	5.5	0.1
2060	5.2	0.3
2065	5.0	0.5
2070	4.4	1.0

TABLE 6-17. X2 MULTIPLIER CHANGE OF POWER LEVEL

Tuned at 2120 mc, 60-mw Input
No Retuning Except Bias Change

Input Power, mw	Output Power, mw	Loss, db
60	27	3.5
50	23	3.4
40	18	3.5
30	13	3.6
20	7	4.6
10	2.4	6.2

TABLE 6-18. X2 MULTIPLIER TEMPERATURE VARIATION

Input Power = 60 mw, f = 2120 mc

Temperature, °C	Output Power, mw
-25	27
0	27
+25	27
+50	27
+75	25
+100	24

TABLE 6-19. X3 MULTIPLIER TEMPERATURE CHECK, S/N 5 MA4078

Tuned at 2006 mc, 10 mw Input, 0-volt Bias (Diode Shorted) +25° C

Power In, mw	+25° C		-20° C		+65° C	
	Out, mw	Loss, db	Out, mw	Loss, db	Out, mw	Loss, db
5	1.32	5.8	0.91	7.4	1.26	6.0
10	3.3	4.8	3.2	5.0	3.1	5.1
15	4.8	4.9	4.9	4.9	4.3	5.4
30	6.8	6.4	7.8	5.9	6.1	6.9

Multiplier was stable over temperature range, -20 to +65° C at all phases of mismatch.

Tuned at 2100 mc, 0.5-volt Bias, 10 mw Input

Power In, mw	+25° C		-20° C		+65° C	
	Out, mw	Loss, db	Out, mw	Loss, db	Out, mw	Loss, db
5	1.05	6.8	0.91	7.4	0.67	8.7
10	3.3	4.8	3.65	4.4	2.0	7.0
15	5.2	4.6	5.85	4.1	3.0	7.0
30	10.0	4.75	10.12	5.5	5.6	7.3

Stable at all phases.

TABLE 6-20. IF PREAMPLIFIER AND POST AMPLIFIER

Temperature: +50°C

Frequency, mc	Input, dbm	Collector Voltage		
		Output		
		-22v, mw	-24v, mw	-26v, mw
78	-80	0.7	0.85	0.95
	78	0.85	0.9	1.05
	76	0.9	1.0	1.1
	74	0.9	1.0	1.1
	72	0.95	1.0	1.1
	70	1.0	1.0	1.1
63	-80	1.0	1.1	1.3
	78	1.2	1.3	1.6
	76	1.4	1.55	1.65
	74	1.55	1.65	1.75
	72	1.55	1.65	1.75
	70	1.55	1.6	1.7
48	-80	0.7	0.75	0.8
	78	0.8	0.9	0.95
	76	0.9	1.0	1.1
	74	1.0	1.1	1.2
	72	1.25	1.35	1.5
	70	1.4	1.55	1.7

TABLE 6-20. (Continued)

Temperature: +22°C

Frequency, mc	Input, dbm	Collector Voltage		
		Output		
		-22v, mw	-24v, mw	-26v, mw
78	-80	0.9	1.0	1.1
	-78	1.0	1.1	1.15
	76	1.05	1.1	1.2
	74	1.1	1.12	1.2
	72	1.1	1.15	1.25
	70	1.1	1.12	1.2
63	-80	0.9	1.1	1.2
	78	1.1	1.3	1.45
	76	1.25	1.4	1.55
	74	1.4	1.5	1.7
	73	1.45	1.5	1.65
	70	1.6	1.55	1.65
48	-80	0.8	0.9	0.95
	78	0.9	1.0	1.1
	76	1.0	1.1	1.2
	74	1.2	1.3	1.4
	72	1.5	1.4	1.6
	70	1.65	1.55	1.75

TABLE 6-20. (Continued)

Temperature: 0°C

Frequency, mc	Input, dbm	Collector Voltage Output -22v, mw	-24v, mw	-26v, mw
78	-80	1.05	1.15	1.25
	78	1.1	1.2	1.3
	76	1.1	1.2	1.3
	74	1.1	1.2	1.25
	72	1.1	1.2	1.25
	70	1.1	1.2	1.25
64	-80	1.2	1.3	1.5
	-78	1.4	1.6	1.7
	76	1.55	1.65	1.8
	74	1.6	1.7	1.8
	72	1.65	1.55	1.75
	70	1.6	1.55	1.75
48	-80	0.75	0.85	0.9
	78	0.9	0.95	1.0
	76	1.0	1.1	1.2
	74	1.15	1.3	1.5
	72	1.3	1.4	1.6
	70	1.5	1.7	1.75

TABLE 6-21. HIGH-LEVEL MIXER

Power Input: Local Oscillator at 25 mw, 4224 mc
Signal at 30 mw, 54 mc

Temperature, °C	Power Output, dbm
-25	5.5
0	5.5
+23	5.4
+50	5.0
+75	4.7

TABLE 6-22. DUAL-FILTER HYBRID (SERIAL NO. 2)

Temperature Environment Data

Temperature °C	Insertion Loss, db: Inputs A to B	Insertion Loss, db: Inputs A to D	Output-to-Output Ratio	Total Insertion Loss, db	VSWR of Terminal A, B, and D	Rejection Input A to Outputs B and D, db: $f_o + 40$ (2152)	Rejection Input A to Outputs B and D, db: $f_o - 40$ (2072)	Isolation Between Outputs B and D, db: $f_o + 40$ (2152)	Isolation Between Outputs B and D, db: $f_o - 40$ (2072)	Hybrid Directivity, db: at f_o (2112)
	$f = f_o = 2112$ mc									
+23	1.9	7.8	5.9	1.0	(A) 1.17 (B) 1.08 (D) 1.08	(B) 45 (D) 50	(B) 44 (D) 52	82	87	22
						(B) 44.4	—	—	—	—
0	1.9	—	—	—	—					
+50	1.9	—	—	—	—	(B) 45.7	—	—	—	—

TABLE 6-23. DUAL SINGLE-SIDEBAND FILTER DIPLEXER (SERIAL NO. 3)

Temperature Environment Data*

Temperature °C	Insertion Loss, db, at f_o (4170)	VSWR at f_o (4170)	Insertion Loss, db, at $f_o + 13$ (4183)	VSWR at $f_o + 13$ (4183)	Insertion Loss, db, at $f_o - 13$ (4157)	VSWR at $f_o - 13$ (4157)	Rejection, db, at 4224
+23	0.60	1.33	1.00	1.43	0.30	1.06	18.3
0	-	-	0.98	-	-	-	17.5
-25	-	-	0.98	-	-	-	16.6
+50	-	-	1.10	-	-	-	19.5
+75	-	-	1.40	-	-	-	20.6

*Data are only from the High-Level Modulator Channel.

TABLE 6-24. ENVIRONMENTAL TEST DATA, 0°C
54 mc IF Wide-Band Limiter

Input, dbm	Output, dbm	Monitor Voltage, volts	Gain Variation at -30 dbm below limiting
0	17.5	-3.00	
- 5	17.0	-2.60	42 mc, 2.7 dbm reference
- 8	16.3	-2.05	54 mc, 2.5 dbm reference
-10	15.5	-1.65	66 mc, 2.0 dbm reference
-12	14.5	-1.30	
-14	13.0	-1.00	at 1 mw input saturation
-16	11.3	-0.760	
-18	9.9	-0.580	42 mc, 2.4 dbm reference
-20	7.8	-0.430	54 mc, 3.1 dbm reference
-22	5.7	-0.340	66 mc, 4.4 dbm reference
-24	3.5	-0.260	
-26	1.4	-0.190	
-28	-0.1	-0.140	
-30	-2.0	-0.110	
-32	-4.0	-0.080	
-34	-6.0	-0.060	
-36	-8.0	-0.045	

Notes:

1) DC input at -24 volts ±1 percent and -4 volts ±1 percent with 1 mw drive — 624 mw

2) Gain variation over band — below limiting 0.7 db; limiting 2.0 db

3) Monitor with 1 mw input and 200 k load — -3.00 volts

TABLE 6-25. ENVIRONMENTAL TEST DATA, 25°C ROOM TEMPERATURE
54 mc IF Wide-Band Limiter

Input, dbm	Output, dbm	Monitor Voltage, volts	Gain Variation at -30 dbm input below limiting
0	17.7	-2.95	
- 5	17.1	-2.60	
- 8	16.5	-2.00	42 mc, 3.2 dbm reference
-10	15.5	-1.60	54 mc, 2.8 dbm reference
-12	14.5	-1.25	66 mc, 2.4 dbm reference
-14	12.9	-1.00	
-16	11.0	-0.760	
-18	9.3	-0.580	
-20	7.5	-0.460	
-22	5.3	-0.340	
-24	3.0	-0.260	
-26	1.0	-0.200	
-28	-1.5	-0.160	
-30	-2.3	-0.130	
-32	-3.9	-0.110	
-34	-6.5	-0.100	
-36	-8.5	-0.080	

Notes:

1) DC input at -24 volts ±1 percent and -4 volts ±1 percent with 1 mw drive — 624 mw

2) Gain variation over band (below limiting) 0.4 db

3) Monitor voltage with 1 mw input and 200 k load — -2.95 volts

TABLE 6-26. ENVIRONMENTAL TEST DATA, 55°C
54 mc IF Wide-Band Limiter

Input, dbm	Output, dbm	Monitor Voltage, volts	Gain Variation
			at -30 dbm below limiting
0	17.7	-3.00	42 mc, 3.5 dbm reference
- 5	17.1	-2.58	54 mc, 3.2 dbm reference
- 8	16.3	-1.95	66 mc, 2.7 dbm reference
-10	15.4	-1.55	at 1 mw input saturation
-12	14.2	-1.20	42 mc, 3.6 dbm reference
-14	12.5	-0.940	54 mc, 2.7 dbm reference
-16	10.8	-0.740	66 mc, 2.2 dbm reference
-18	9.0	-0.580	
-20	7.0	-0.460	
-22	5.0	-0.365	
-24	2.7	-0.310	
-26	0.8	-0.280	
-28	-1.7	-0.265	
-30	-2.6	-0.260	
-32	-4.7	-0.260	
-34	-6.8	-0.260	
-36	-8.9	-0.260	

Notes:

1) DC input at -24 volts ± 1 percent and -4 volts ± 1 percent with 1 mw drive — 624 mw

2) Gain variation over band — below limiting 0.8 db; limiting 1.4 db

3) Monitor with 1 mw input and 200 k load — -3.00 volts

Table 6-27. ENVIRONMENTAL TEST DATA, -25°C
54 mc IF Wide-Band Limiter

Input, dbm	Output, dbm	Monitor Voltage, volts	Gain Variation
			At -30 dbm below limit
0	17.2	3.00	
- 5	16.8	2.50	42 mc 2.2 dbm Ref.
-10	15.5	1.60	54 mc 3.4 dbm Ref.
-12	14.5	1.25	66 mc 1.6 dbm Ref.
-14	13.3	0.950	
-16	11.5	0.725	At 1 mw input saturation
-18	9.0	0.560	
-20	7.0	0.420	42 mc 4.5 dbm Ref.
-22	6.0	0.320	54 mc 3.2 dbm Ref.
-24	3.8	0.225	66 mc 2.5 dbm Ref.
-26	1.5	0.160	
-28	0	0.120	
-30	- 2.8	0.080	
-32	- 3.7	0.060	
-34	- 5.8	0.040	
-36	- 6.5	0.030	

Notes:

1) DC input at -24 volts ±1 percent and -4 volts ±1 percent with 1 mw drive — 624 mw

2) Gain variation over band — below limiting 1.8 db; limiting 2.0 db

3) Monitor with 1 mw input and 200 k load — -3.00 volts

Table 6-28 . ENVIRONMENTAL TEST DATA, 75°C
54 mc IF Wide-Band Limiter

Input, dbm	Output, dbm	Monitor Voltage, volts	Gain Variation At -30 dbm below limit
0	17.9	-3.00	
- 5	17.4	-2.60	42 mc 3.5 dbm Ref.
- 8	16.5	-2.00	54 mc 3.1 dbm Ref.
-10	15.5	-1.60	66 mc 3.0 dbm Ref.
-12	14.5	-1.28	
-14	12.5	-1.05	At 1 mw input saturation
-16	10.9	-0.820	
-18	9.1	-0.710	42 mc 3.4 dbm Ref.
-20	7.8	-0.680	54 mc 2.6 dbm Ref.
-22	5.0	-0.668	66 mc 2.1 dbm Ref.
-24	2.9	↑	
-26	1.0	levels	
-28	- 0.7	off	
-30	- 2.5	remains	
-32	- 4.6	0.660	
-34	- 6.6		
-36	- 8.8	↓	

Notes:

1) DC input at -24 volts ±1 percent and -4 volts ±1 percent with 1 mw drive — 624 mw

2) Gain variation over band — below limiting 0.5 db; limiting 1.3 db

3) Monitor with 1 mw input and 200 k load — -3.00 volts

Table 6-29 . SINGLE-SIDEBAND FILTER (f_o = 2085 mc)(SERIAL NO. 1)

Temperature Environment Data

Temperature °C	Insertion Loss, db, at f_o (2085)	VSWR at f_o (2085)	Insertion Loss, db, at $f_o + 8$ (2093)	VSWR at $f_o + 8$ (2093)	Insertion Loss, db, at $f_o - 8$ (2077)	VSWR at $f_o - 8$ (2077)	Rejection, db at $f_o + 55$ (2140)	Rejection, db at $f_o + 55$ (2130)
+23	0.55	1.60	0.70	1.60	0.80	1.53	27.5	31.5
0	----	----	0.65	----	----	----	27.0	----
+50	----	----	0.80	----	----	----	28.1	----

Table 6-30 . SINGLE-SIDEBAND FILTER (f_o = 2119 mc)(SERIAL NO. 3)

Temperature Environment Data

Temperature °C	Insertion Loss, db, at f_o (2119)	VSWR at f_o (2119)	Insertion Loss, db, at $f_o + 3$ (2122)	VSWR at $f_o + 3$ (2122)	Insertion Loss, db, at $f_o - 3$ (2116)	VSWR at $f_o - 3$ (2116)	Rejection, db at $f_o + 66$ (2185)	Rejection, db at $f_o + 66$ (2053)
+23	1.15	1.15	1.25	1.23	1.20	1.22	55.0	55.0
0	----	----	1.15	----	----	----	54.5	----
+50	----	----	1.35	----	----	----	55.8	----

Table 6-31 . BANDPASS CHARACTERISTIC OF X32 MULTIPLIER AND PHASE MODULATOR COMBINATION

Room Temperature (24°C)

1) Carrier power as a function of supply voltage.

Supply voltage, volts	-22	-23	-24	-25	-26
Carrier power, milliwatts	11	13	15	16	17

2) Bandpass characteristic at various supply voltages. Sideband power in decibels with respect to 15 mw (+12 dbm).

Modulation Frequency, mc	-22 volts	-23 volts	-24 volts	-25 volts	-26 volts
33.0	-25	-25	-25	-25	-26
33.5	-25	-25	-25	-26	-25
34.0	-25	-25	-25	-26	-25
34.5	-26	-25	-24	-26	-24
35.0	-26	-25	-24	-26	-23
35.5	-26	-25	-24	-26	-23
36.0	-25	-25	-26	-26	-23
36.5	-23	-25	-26	-27	-25
37.0	-20	-24	-27	-29	-27
37.5	-25	-21	-27	-32	-33
38.0	-34	-27	-28	-34	-35
38.5		-34	-29	-34	-36
39.0	-40	-39	-32	-34	-36
39.5					
40.0	-41	-44	-43	-39	-39

TABLE 6-31. (continued)

Low Temperature (0° C)

1) Carrier power as a function of supply voltage.

Supply voltage, volts	-22	-23	-24	-25	-26
Carrier power, milliwatts	12	14	16	17	20

2) Bandpass characteristic at various supply voltages.
Sideband power in decibels with respect to 15 mw (+12 dbm).

Modulation Frequency, mc	-22 volts	-23 volts	-24 volts	-25 volts	-26 volts*
33.0	-25	-25	-24	-26	
33.5	-26	-25	-24	-27	
34.0	-26	-25	-24	-27	
34.5	-27	-26	-23	-27	
35.0	-26	-27	-24	-26	
35.5	-26	-27	-25	-25	
36.0	-26	-27	-26	-26	
36.5	-25	-27	-27	-27	
37.0	-24	-27	-28	-29	
37.5	-26	-23	-28	-33	
38.0	-33	-24	-29	-36	
38.5	-39	-32	-30	-38	
39.0	-41	-41	-33	-38	
39.5	-41	-47	-38	-40	
40.0	-41	-47	-42	-43	

*Oscillation occurs under these conditions. The oscillation is well below the carrier level but is only a few decibels below the signal level.

TABLE 6-31. (continued)

High Temperature (50°C)

1) Carrier power as a function of supply voltage.

Supply voltage, volts	-22	-23	-24	-25	-26
Carrier power, milliwatts	7	9	11	12	13.5

2) Bandpass characteristic at various supply voltages.
Sideband power in decibels with respect to 15 mw (+12 dbm).

Modulation Frequency, mc	-22 volts	-23 volts	-24 volts	-25 volts	-26 volts
33.0	-26	-25	-25	-24	-24
33.5	-26	-25	-25	-24	-24
34.0	-27	-25	-25	-24	-24
34.5	-27	-26	-24	-24	-24
35.0	-26	-26	-24	-24	-23
35.5	-26	-26	-26	-25	-23
36.0	-24	-26	-27	-27	-25
36.5	-24	-25	-27	-29	-28
37.0	-26	-25	-27	-30	-30
37.5	-31	-26	-26	-29	-30
38.0	-35	-30	-26	-27	-29
38.5	-39	-35	-27	-24	-27
39.0	-39	-38	-31	-22	-23
39.5	-40	-40	-37	-28	-17
40.0	-40	-40	-40	-34	-25
40.5	-	-	-	-	-32
41.0	-	-	-	-	-37

TABLE 6-32. SAMPLE CALCULATIONS

1) Calculation of sideband level in dbm, from data taken in db below 15 mw.

At room temperature and modulation frequency of 33.0 mc at V_{supp} = -24 volts the measured sideband power is -25 db with respect to 15 mw.

$$10 \log_{10} \frac{15 \text{ mw}}{1 \text{ mw}} - 25 \text{ db} = P_{dbm}$$

$$+ 12 \text{ dbm} - 25 \text{ db} = -13_{dbm}$$

2) Calculation of frequency of sideband from modulation frequency.

Sideband frequency = carrier frequency + modulation frequency - 32.5 mc.

For modulation frequency of 33 mc

$$f_{S.B.} = f_c + 33.0 - 32.5 = f_c + 0.5 \text{ mc}$$

Transponder System Components

Components Common to Both Transponders

X3 Multiplier. The design of this unit is continuing with further development of input circuitry. It appears that two models will be necessary to cover the four frequency channels.

X2 Multiplier. A new breadboard model is being fabricated to utilize a different feed-through capacitor. This design has been finalized and product design started. One model will be used for all frequencies.

6-gc Ferrite Switch. The 4-gc ferrite switch is being scaled in frequency to produce this unit. The design is progressing and no problems are anticipated.

6-gc Mixer. This unit is undergoing minor modification to reduce the capacity of the IF output. The 6-gc local oscillator input circuitry is being slightly modified to eliminate a spurious mode.

X32 Multiplier. It has been decided that one basic model will be utilized to cover all frequency bands. Applicable tuning adjustments for channel coverage is being completed.

Frequency Translation Transponder Components

64-mc IF Preamplifiers. These units have been modified to operate at the new IF frequency of 64 mc. A noise figure of 3.5 db has been measured, which is acceptable.

64-mc IF Limiter. This unit has also been changed to 64-mc operation. It has been redesigned so that a provision for inserting the beacon signal is now included.

Dual-Single Sideband Filter-Diplexer (4080-4170 mc),

Beacon Mixer and 14-db Coupler. These units have been eliminated, since the beacon signal is now being inserted at IF in the IF limiter.

High-Level Mixer. A varactor type mixer is now under development to replace the varistor type mixer (Figure 6-14). A lower loss mixer will result from the new design.

Multiple Access Transponder Components

30-mc IF Phase Modulator. The breadboard of this unit has been completed.

Receiver Master Oscillator (66.223 mc) and Transmitter Master Oscillator (32.5 mc). These units have been replaced by a single oscillator (32 mc) and a doubler (Figure 6-15).

Filter Amplifier (33 to 38 mc). A new unit, a filter amplifier, has been designed and breadboarded to eliminate an image response in the IF phase modulator and thereby reduce the noise in the multiple access transponder system.

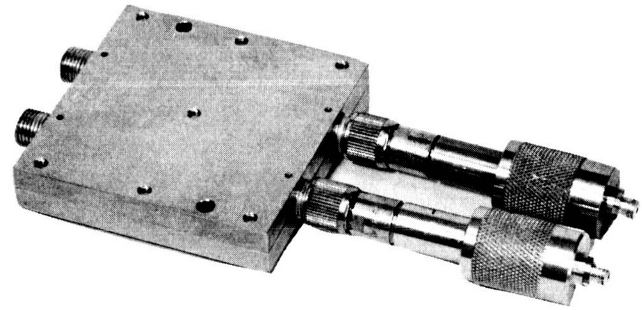


Figure 6-14. Varistor-Type Mixer

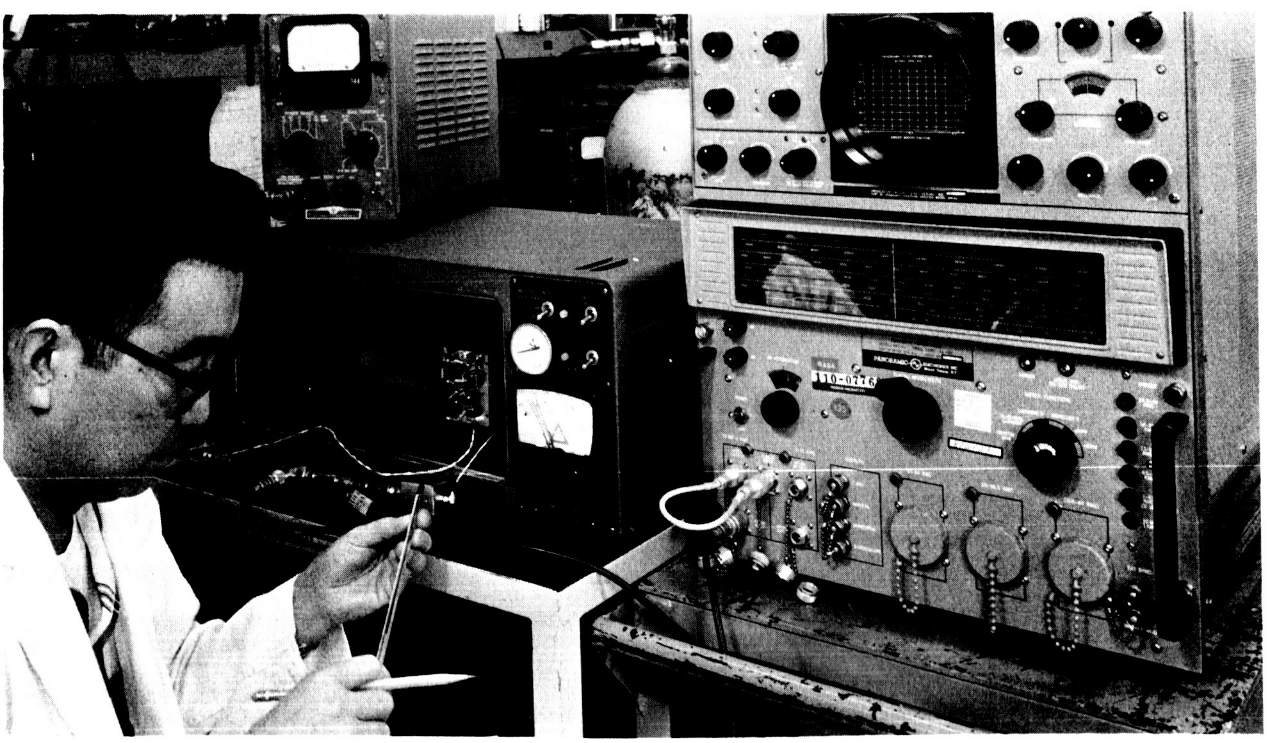


Figure 6-15. Temperature Testing of Master Oscillator

Advanced Syncom Support Transponder Specification

1.0 SCOPE

1.1 Scope: This specification covers the minimum operational requirements of the Advanced Syncom support transponder.

2.0 OBJECTIVE

2.1 Objective: The primary objective of the support transponder is to check out and validate the Syncom satellite ground stations by simulating vehicle transponder transmission and reception.

3.0 CONFIGURATION

3.1 Configuration: The support transponder shall consist of two separate units--the transponder package and the traveling-wave tube (TWT) package.

3.2 Transponder Package: The transponder package shall consist of the following units:

3.2.1 Frequency translation transponder

3.2.2 Multiple access transponder

3.2.3 Power supply and voltage regulators

3.2.4 Input and output signal attenuators

3.2.5 Switching panel

3.2.6 Remote switching panel

3.2.7 Interconnecting cables

3.3 TWT Package: The TWT package shall consist of the following units:

3.3.1 TWT

3.3.2 TWT power supply

4.0 TRANSPONDER PACKAGE CHARACTERISTICS

4.1 Operating Frequencies: The operating frequencies shall be:

4.1.1 The operating input frequency shall be one of four operating frequencies assigned to Syncom II and shall be specified by the user.

4.1.2 The operating output frequency shall be compatible with the input frequency of 4.1.1.

4.2 Frequency Translation Mode: The frequency translation unit shall be provided to translate and amplify the signal carrier frequency with no conversion in modulation.

4.2.1 RF Bandwidth: The 3-db bandwidth shall be 25 ±1.5 mc measured between the IF input and RF output.

4.2.2 Noise Figure: The noise figure shall be better than 9 db referenced to the standard noise temperature of 290° K.

4.2.3 Frequency Stability: The beacon signal frequency shall be stable to within 0.002 percent. The stability of the other output frequencies shall be consistent with the beacon signal stability.

4.2.4 Power Output: The power output shall be at least 1 milliwatt into a 50-ohm load.

4.3 Multiple Access Mode: The multiple access transponder shall be provided to convert the single-sideband signals into phase-modulated signals and to translate the frequency up to the proper value.

4.3.1 RF Bandwidth: The 3-db bandwidth for the single-sideband up link shall be 5 mc (±1 mc - 0.5 mc).

4.3.2 Noise Figure: The noise figure of the unit shall be better than 9 db referenced to the standard noise temperature of 290° K.

4.3.3 Capacity: The multiple access transponder shall be able to convey up to 1200 one-way, 4-kc voice channels.

4.3.4 Power Output: The power output shall be at least 1 milliwatt into a 50-ohm load.

4.4 Power Supply Requirements

4.4.1 The power supply shall consist of the following:

4.4.1.1 One -28 volt dc supply capable of operating from 115 volts ac, 60 cps.

4.4.1.2 One battery pack capable of providing operating power for 1 hour.

4.4.1.3 Frequency Translation Regulator

4.4.1.4 Multiple Access Regulator

4.4.1.5 Provisions for operating from an external -28 volt dc source.

4.4.2 Voltage: The power supply voltage shall be within -26 and -35 volts dc during normal operating conditions.

4.4.3 Transient Stability: The power supply voltage shall remain within the -25 to -35 volt dc range during any transfer of equipment loads from any mode of operation and shall recover and remain within the steady-state limits within 0.5 second.

4.4.4 Ripple: The peak-to-peak ripple voltage produced by the power supply, when measured by a VTVM in series with a 0.4 microfarad capacitor shall not exceed 0.5 volt.

4.4.5 Electrical Loads: The power supply shall be capable of supplying power for the following equipment:

4.4.5.1 Frequency translation transponder

4.4.5.2 Multiple access transponder

4.4.5.3 TWT power supply

4.4.5.4 Indicator lamps

4.4.6 Battery Pack: The battery pack shall consist of rechargeable nickel-cadmium cells capable of providing operating power for 1 hour.

4.4.6.1 Recharging: The batteries shall be capable of being recharged from the ac power supply in the support transponder.

4.4.7 Regulators: Two voltage regulators shall provide -24 volts dc ±1 percent to the frequency translation transponder and the multiple access transponder.

4.4.8 Overload Protection: A fuse shall be provided in both the ac and dc input power lines to the support transponder.

4.5 Front Panel: The following is a list of controls, meters, etc., which will be available on the front panel of the support transponder.

4.5.1 Switches: The following switches shall be on the front panel:

4.5.1.1 External ac power, ON-OFF

4.5.1.2 External dc power, ON-OFF

4.5.1.3 Power selector (external-internal)

4.5.1.4 Battery charge, ON-OFF

4.5.1.5 Frequency translation transponder on multiple access off

4.5.1.6 Multiple access transponder on frequency translation off

4.5.1.7 TWT filament, ON-OFF

4.5.1.8 TWT high voltage, ON-OFF

4.5.1.9 Input signal attenuator, 0-100 db in 10 db steps

4.5.1.10 Input signal attenuator, 0-10 db in 1 db steps

4.5.1.11 Output signal attenuator, 0-100 db in 10 db steps

4.5.1.12 Output signal attenuator, 0-10 db in 1 db steps

4.5.2 Lights: The following lights shall be on the front panel:

4.5.2.1 External ac power, ON

4.5.2.2 External dc power, ON

4.5.2.3 Frequency translation transponder, ON

4.5.2.4 Multiple access transponder, ON

4.5.2.5 TWT filaments, ON

4.5.2.6 TWT high voltage, ON

4.5.3 Meters: Meters shall be provided to monitor the following:

4.5.3.1 Unregulated power supply voltage

4.5.3.2 Battery charging current

4.5.3.3 Power supply current

4.5.4 Connectors: The following connectors shall be on the front panel:

4.5.4.1 External ac power input

4.5.4.2 External dc power input

4.5.4.3 Signal input to attenuator (BNC)

4.5.4.4 Signal output from attenuator (BNC)

4.5.4.5 TWT power supply connector (Cannon)

4.5.4.6 Remote switching panel connector (Cannon)

4.5.5 Fuses:

4.5.5.1 External ac power

4.5.5.2 External dc power

4.5.6 Remote Switching: A remote switching panel with appropriate cables shall be provided to operate the support transponder from a distance of 200 feet.

4.6 RF Attenuators: RF attenuators for input and output signals shall be provided in the support transponder.

4.6.1 Input Attenuator: 0-100 db attenuation, selected in 1 db steps, shall be provided to receive input signals of up to -20 dbm.

4.6.2 Output Attenuator: 0-100 db attenuation, selectable in 1 db steps, shall be provided for the output signal.

4.7 Cooling and Heating: The support transponder shall be capable of operating from -10 to +120° F.

4.7.1 AC Blower: An ac blower capable of providing adequate cooling when operating from a 60 cps ac source shall be provided.

4.7.2 AC Heater: An ac heater capable of providing adequate heating when operating from a 60 cps ac source shall be provided.

4.7.3 DC Blower: A dc blower capable of providing adequate cooling when operating from an external -28 volt dc source shall be provided.

5.0 TWT PACKAGE CHARACTERISTICS

5.1 Input Requirements

5.1.1 Input Power: The input power level shall be at least 1 milliwatt.

5.1.2 Input Frequency: The input frequency shall be one of four to be specified by the customer and compatible with 4.1.2.

5.2 Output Characteristics

5.2.1 Output Power: The RF output power shall be at least 3.9 watts.

5.2.2 Output Frequency: The output frequency shall be as specified in 4.1.2.

5.2.3 Filter: An S-band output filter shall be provided.

5.3 Power Supply

5.3.1 Input Power: The input power shall be provided from the transponder package.

5.3.2 Internal Power: A TWT internal power supply which includes a regulator shall be provided for the TWT filaments and high voltage.

5.4 Monitoring Provisions

5.4.1 Outputs: A 20-db directional coupler shall be provided at the output of the TWT for power measurements.

5.5 Heating and Cooling

5.5.1 The heating and cooling shall be the same as specified in section 4.7.

5.6 Front Panel

5.6.1 Connectors: The following connections shall be on the front panel:

5.6.1.1 Input power connector (from transponder package).

5.6.1.2 Input signal connector.

5.6.1.3 Output signal connector from directional coupler.

5.6.1.4 Power out connection from 20-db down directional coupler.

TRAVELING-WAVE TUBE POWER AMPLIFIER

During this report period six tubes (No. 25 to 30) with the smaller diameter helix design were constructed. Although some of these tubes have not been tested, an optimum helix pitch has been obtained for the small diameter helix for the 4.0 watts, 30 percent efficiency, and 36 db saturated gain specifications. Present experimental data indicate that this design is superior in many respects to the larger diameter helix as used in tube 384H-13.

Status of Six Additional TWTs Fabricated for Test Purposes

A summary of the characteristics of the last eight tubes is given in Table 6-33 This list is a continuation of the list given in the Summary Report and includes the test results of two tubes (No. 23 and 24) that were on the previous list but without test results.

Evaluation of Alternate TWT Designs

Although the performance of the design used on tubes No. 25, 26, and 28 was satisfactory, the design incorporated in tube No. 27 appears to be optimum for the present performance requirements. The major differences between these two designs is in the helix pitch; all of these tubes have the small diameter helix design. Tube No. 27 has lower anode voltage, higher beam voltage, less power variation with beam voltage fluctuations, bandwidth centered higher in frequency, and slightly lower basic efficiency and gain than previous tubes. As Table 6-33 shows, a large percentage of tubes, although not optimum in design, meet specifications. These tubes will be placed in storage for possible future testing.

The performance characteristics and operation parameters of tube No. 27 before packaging were:

Performance Characteristics:

Frequency, kmc	3.9	4.0	4.2
P_{out}, watts	4.17	4.35	4.30
P_{in}, mw	0.80	0.80	0.80
Gain at P_{in} = 0.80 mw, db	37.1	37.3	37.3
Total efficiency (including heater power), percent	31.6	32.4	31.8

TABLE 6-33. TRAVELING-WAVE TUBE CHARACTERISTICS

Tube No.	Major Changes from Previous Tubes	Some Test Results	Status
23	Modification of No. 20 to raise power	Tube meets specifications. During bakeout a leak in the output ceramic was discovered. The leak was repaired with gliptol compound which makes the tube suitable only for RF tests.	Storage
24	Similar to No. 23	Construction error prevented any useful experiments.	Scrapped
25	Modification of No. 24 to raise beam voltage	Tube meets specifications before packaging.	Being packaged
26	Same as No. 25	Tube meets specifications before packaging.	Awaiting test after packaging
27	Modification of No. 26 to raise beam voltage	Before packaging, tube met specifications. Anode voltage was under 200 volts and tube appeared less sensitive than previous designs to changes in beam voltage.	Awaiting test after packaging
28	Same as No. 26	Initial data indicates gun perveance is low.	Still undergoing test
29	Same as No. 27 except for a minor modification of the loss distribution		On bakeout
30	Same as No. 27 except for a minor modification of the loss distribution		On bakeout

Operation Parameters:

Element	Voltage	Current
Cathode	-1300	19.3 ma
Anode	+125	Negligible
Helix	0	1.70 ma
Collector	-725	17.6 ma
Heater	4.5	0.260 amp

Some characteristics of tube No. 27 are shown in Figure 6-16. Figure 6-16a shows that the power output with a power input of 0.80 mw is a maximum at 4.1 kmc and the tube delivers 4.0 watts output from 3.85 to 4.4 kmc. Data on tube No. 13 which is undergoing qualification testing is shown in Figure 6-16a for comparison. Figure 6-16b plots beam efficiency as a function of frequency for depressed collector operation, and Figure 6-16c illustrates the broad-band saturated gain characteristics.

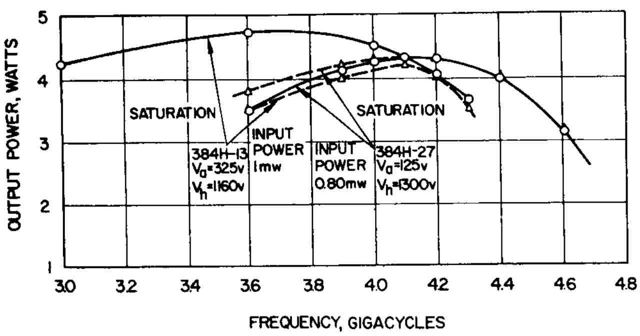
The results of current testing have shown the small diameter helix design to be superior to the larger diameter helix design in many respects. The most important is the bandwidth location. As Figure 6-16a illustrates, the center frequency of the large helix design (tube No. 13) is about 3.5 kmc and the center of the small helix design is about 4.1 kmc. An important advantage of operation at midband is less variation in power output with fluctuations in beam voltage and power input. The reduced helix diameter has not affected the excellent focusing of the original design. The beam transmission of 384H-27 with RF was 91 percent with a collector potential depression of 55 percent. The high anode voltage (375 volts) of the large diameter helix design has been reduced to less than 200 volts with the smaller helix.

As indicated in the tube list, more tubes like No. 27 are being made to verify the performance of the small diameter helix design. Future tubes will be made slightly longer to increase the gain to over 40 db.

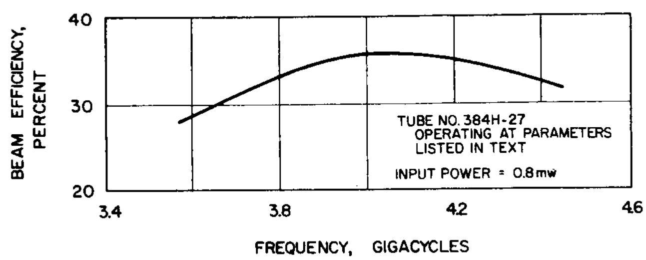
Qualification Testing of Traveling-Wave Tube

A Syncom II traveling-wave tube, MTD 384H-No. 13, is being subjected to qualification test requirements with all tests complete except the 7-day spin test. The tube was completely checked before and after each of the tests. The tests were performed in the following order:

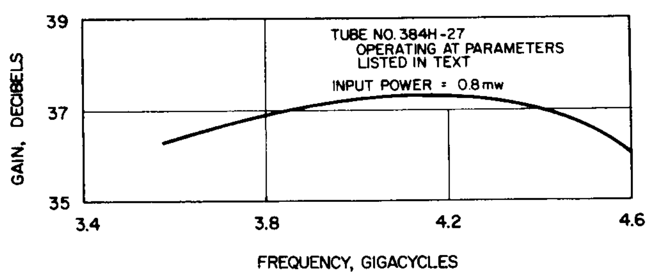
1) Thermal Vacuum. The TWT was subjected to a thermal vacuum test as outlined below. The tube was attached to a mounting plate and the surface mounting temperature was varied as



a) Output power versus frequency



b) Beam efficiency versus frequency



c) Gain versus frequency

Figure 6-16. Frequency Graphs

required. The TWT was operated continuously on dc power and the RF amplification was checked out every 24 hours.

Pressure	Mounting Surface Temperature	Test Duration
1 x 10-5 torr	25 to 35° F	72 hours
1 x 10-5 torr	165 to 175° F	72 hours

After thermal vacuum test, all parameters of the TWT were checked and no changes were noted because of the test.

2) Vibration. The TWT was subjected to the vibration environments in each of the three orthogonal axes as indicated below. The TWT was checked at the completion of each frequency sweep.

Vibration Levels

a) Sinusoidal (logarithmic sweep)

Frequency, cps	Thrust Axis (g, O-peak)	Time, minutes
5-15	0.25 inch DA	0.75
15-60	0.5 inch DA to 24.2, 15 g to 60 cps	1.00
60-100	30	0.38
100-250	15	0.63
250-400	10	0.38
400-2000	7.5	1.20

Frequency, cps	Transverse Axis (g, O-peak)	Time, minutes
5-15	0.25 inch DA	0.75
15-25	0.5 inch DA	0.38
25-60	0.5 inch DA to 35 then 30 g to 60 cps	0.63
60-250	15	1.00
250-400	10	0.38
400-2000	7.5	1.00

b) Random

Frequency, cps	All Axes (PSD)	Time, minutes
20-80	Flat 0.04 g 2/cps	6
80-1280	Increasing from 0.04 g 2/cps to 0.07 g 2/cps at 1.22 db/octane	6
1280-2000	0.97 g 2/cps	6

After vibration tests, all parameters of the TWT were checked and no changes were noted because of test.

3) Shock and Sustained Acceleration

Shock: The TWT was subjected to a sine acceleration pulse in each of the three orthogonal axes as indicated below. The tube was checked after each test.

Direction	Acceleration, g	Duration, milliseconds
Forward	30	11
Lateral	15	11

Acceleration: The TWT was subjected to an acceleration environment as shown below. The tube was checked out after each test.

Axis	Acceleration, g	Duration, minutes
Thrust (Z-Z)	+30	10
Thrust (Z-Z)	-30	10
Transverse (X-X)	+6	4
Transverse (X-X)	-6	4
Transverse (y-y)	+6	4
Transverse (y-y)	-6	4

After these tests, all parameters of the TWT were checked and no changes were noted because of the test.

4) Spin. The TWT will be subjected to a spin test as outlined below. The tubes will be oriented on the spin fixtures in a position approximating the mounting angle on the spacecraft. The TWT will be operated continuously on dc power.

Acceleration	Rotational Speed	Time
12 g	150 rpm	7 days

Fabrication and Test of Breadboard TWT Power Supply

The breadboard TWT power supply (Figure 6-17) was reported in the Syncom II Summary Report. However, certain changes are contemplated which have not been tested because the parts necessary were not yet available. Those areas where changes are anticipated, and the reasons for the changes, are indicated below.

High-Voltage DC-DC Converter

Transformer. The high-voltage transformer tested was identical to the Syncom I transformer except that the turns ratios were altered to accommodate the higher voltages of the Syncom II TWT, and the collector-cathode output winding wire size was changed from AWG 37 to AWG 40. The smaller diameter wire was necessary to allow the increased number of turns to fit on the same magnetic core. This essentially doubled the inherent voltage regulation of the collector-cathode output, increasing it to 2.4 percent. Since the cathode-collector load is quite constant, this increased regulation is not considered detrimental. However, the current density has increased from 1000 to 600 circular mils/amp. The increased temperature rise is yet to be determined with the 475175-100 unit thermal environment simulated. It may be necessary to change the core to one having a slightly larger window area. To use AWG 37 wire on the collector-cathode winding, the transformer volume and weight would have to increase approximately 15 percent. This would reduce the voltage regulation and current density to the Syncom I converter level.

High-Voltage Diodes and Filters. The diodes used in the Syncom I high-voltage outputs have proved satisfactory for Syncom II, except that two diodes in series per leg of the helix-collector and collector-cathode bridges were required. This was necessitated by the higher TWT voltages. Remaining to be evaluated are diodes of the type used in the Surveyor TWT power supplies. The latter are higher voltage units, having a faster recovery time. This can result in lower ripple for the existing output filters, or smaller output filters for the existing ripple specifications. Assuming that the Syncom I ripple specifications will also apply to Syncom II, the high-voltage output filtering will be changed if the faster diodes are selected. The same

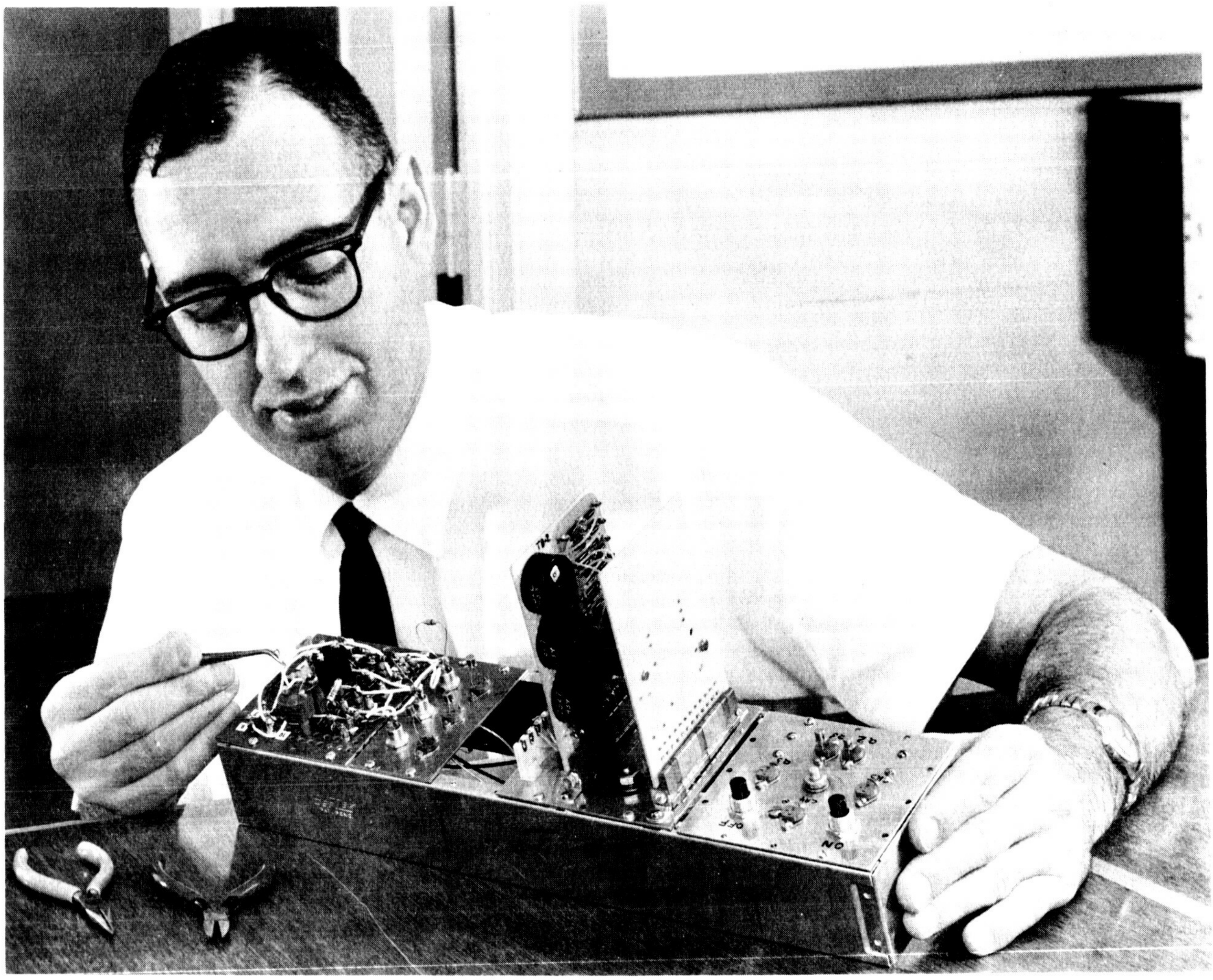


Figure 6-17. Traveling-Wave Tube
Power Supply Breadboard

chokes would probably be retained, with reduced capacitance. Smaller values of capacitance would be most desirable since higher voltage capacitors must now be used.

Switching Transistors. The 2N1724 transistors employed in Syncom I performed very well. However, they were much larger than necessary. This selection resulted from an efficiency versus weight tradeoff early in the Syncom I program. Otherwise, the 2N2151 transistor would have been selected. The result was a 2 percent increase in efficiency which reduced the overall losses 0.27 watt. However, the transistor weights were 15.5 grams for each 2N1724 and 6.0 grams for each 2N2151.

To achieve an efficiency of approximately 90 percent for the Syncom I high-voltage converters, it was necessary to hand-select the base drive current limiting resistor for each converter to match the H_{FE} variations of the transistors. This resulted in a rather cumbersome fabrication procedure. It is anticipated that a larger unregulated bus capacity will allow a relaxation of Syncom II efficiency specifications. A weight saving of 25 grams per 475-unit plus the elimination of one selected resistor can be achieved at the cost of approximately 0.5 watt. The decision is yet to be made relative to this tradeoff.

Filament Inverters

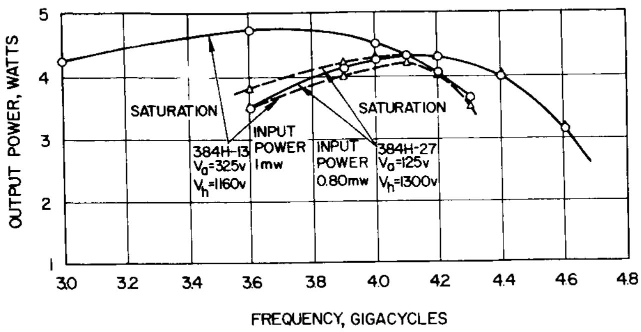
The constant power filament supply was evaluated using 2N2151 switching transistors because they were readily available. This is a stud-mounted device, and is larger than required for the power involved. 2N1717 transistors (T.O 5 package) will be evaluated as soon as they are received. The 84 percent efficiency reported before may be somewhat optimistic. However, 80 percent should be easily achieved.

PHASED ARRAY ANTENNA

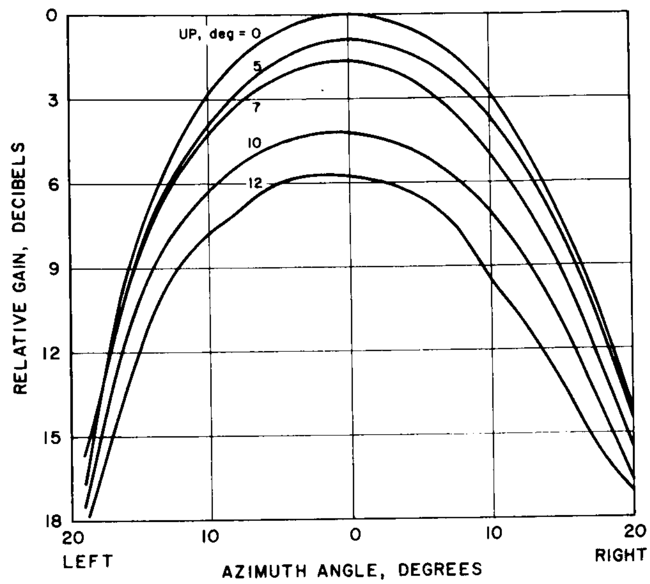
Antenna Patterns on Existing Phased Array

Figure 6-18 contains plots of the phased array antenna pattern measured in the main beam. The tests were made in the laboratory with the antenna mounted on a rotatable table about half-way from floor to ceiling. Microwave absorbent material was placed behind the phased array to reduce backlobe radiation and consequent reflections off the walls of the room. A horn antenna used as a receiver was mounted on a pole, so that its height could be varied, and was maintained at 11 feet from and pointing toward the phased array. The "demonstration" control electronics system was used to drive the ferrite phase shifters.

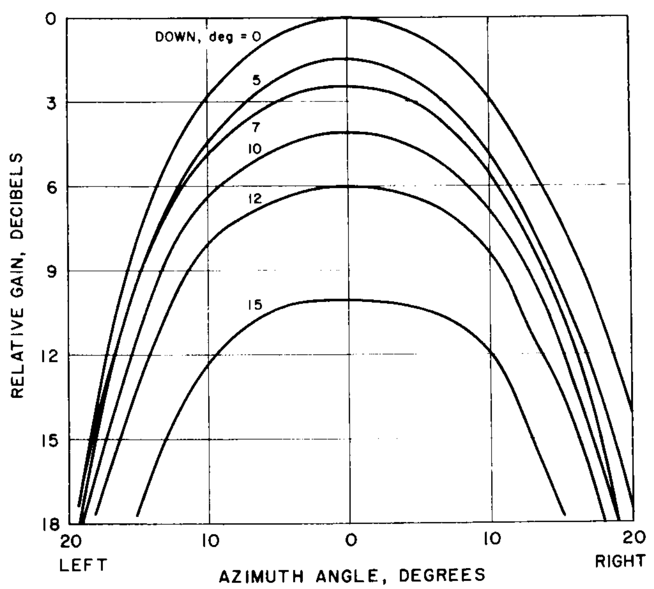
Data were obtained for azimuth angles (around the spacecraft spin axis) of ±20 degrees and elevation angles from 12 degrees up to 15 degrees down (down referring to the forward or apogee engine end of the spacecraft). The data were obtained on the basic array, and also with a 30-inch-diameter



a) 0° azimuth

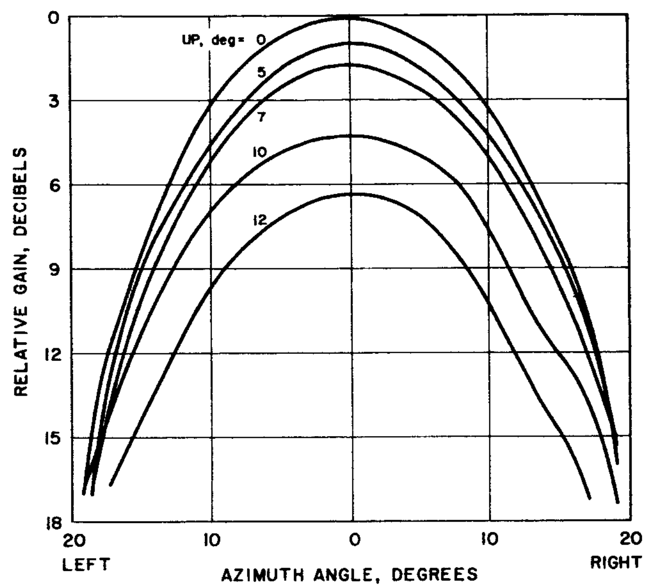


b) No ground plane, 4050 mc, up-aft

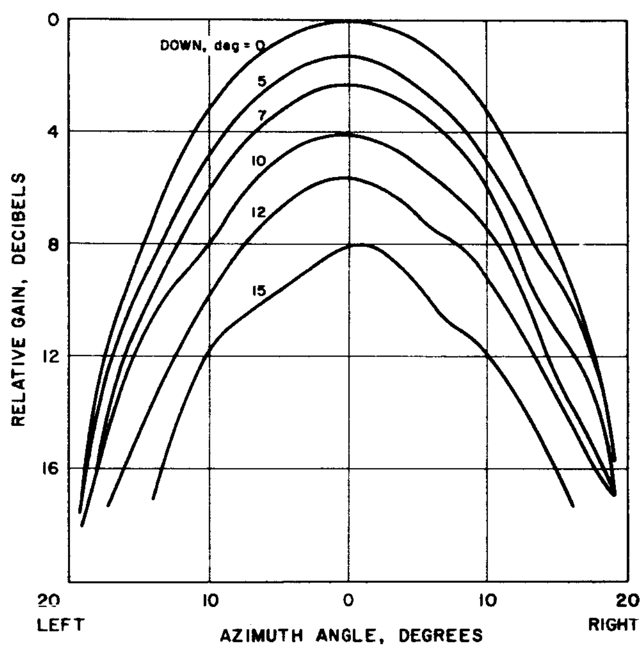


c) No ground plane, 4050 mc, down-forward

Figure 6-18. Phased Array Vertical Pattern



d) 30-inch diameter ground plane, 4050 mc, up-aft



e) 30-inch diameter ground plane, 4050 mc, down-forward

Figure 6-18 (continued). Phased Array Vertical Pattern ground plane located at the base of the array, about 5 inches below the lowest radiating element. The results were approximately the same, although the ground plane had a minor effect in the down direction. All data were obtained at 4050 mc, the frequency at which the antennas were matched. The horizontal beam width was approximately 20.5 degrees while the elevation beam width was approximately 17.6 degrees. Elevation coverage was limited by the height of the room, while the azimuth data were limited by fluctuations in the readings for low signals. A repeat of these data will be made on an open antenna range to check validity of the laboratory measurements.

Stripline Design

The output-coupler stripline design layouts have been completed, and the inked layout artboard for one of the two is also finished. The basic ground plane layout is finished and the drawings have been submitted for fabrication. Four of these ground planes are used in the output coupler and two in the input power splitter; they are basically the same except for screw-hole locations.

The layout for the power splitter stripline is also completed. It is based on a new design hybrid ring whose characteristics are shown in Figure 6-19. Over the normal operating band of about 3980 to 4200 mc its operation is excellent.

PHASED ARRAY CONTROL ELECTRONICS

PACE Circuitry

The circuits and subsystems described in the Summary Report have been under development. The only problem to date has been in designing the simplified integrator. The present design is adequate for PACE usage, but lacks sufficient accuracy to calculate the ψ-ψ_2 angle to 0.1 degree. Computational data will be available to show whether or not sufficient accuracy can be achieved with the simple integrator to justify an on-board computation of the ψ-ψ_2 angle. The extra effort on the integrator will provide additional margin for the PACE system.

PACE operation is essentially that described in the Summary Report. However, the system has been modified to permit operation during eclipse periods. Just prior to entering the eclipse, a command would be sent which switches the PACE input from the solar sensor to a command source. Pseudo ψ pulses would then be transmitted from the ground.

During the period since publication of the Summary Report, the semiconductor specifications for the digital equipment have been in preparation. This involved evaluation of new devices, generation of specifications for their use, and upgrading of existing specifications.

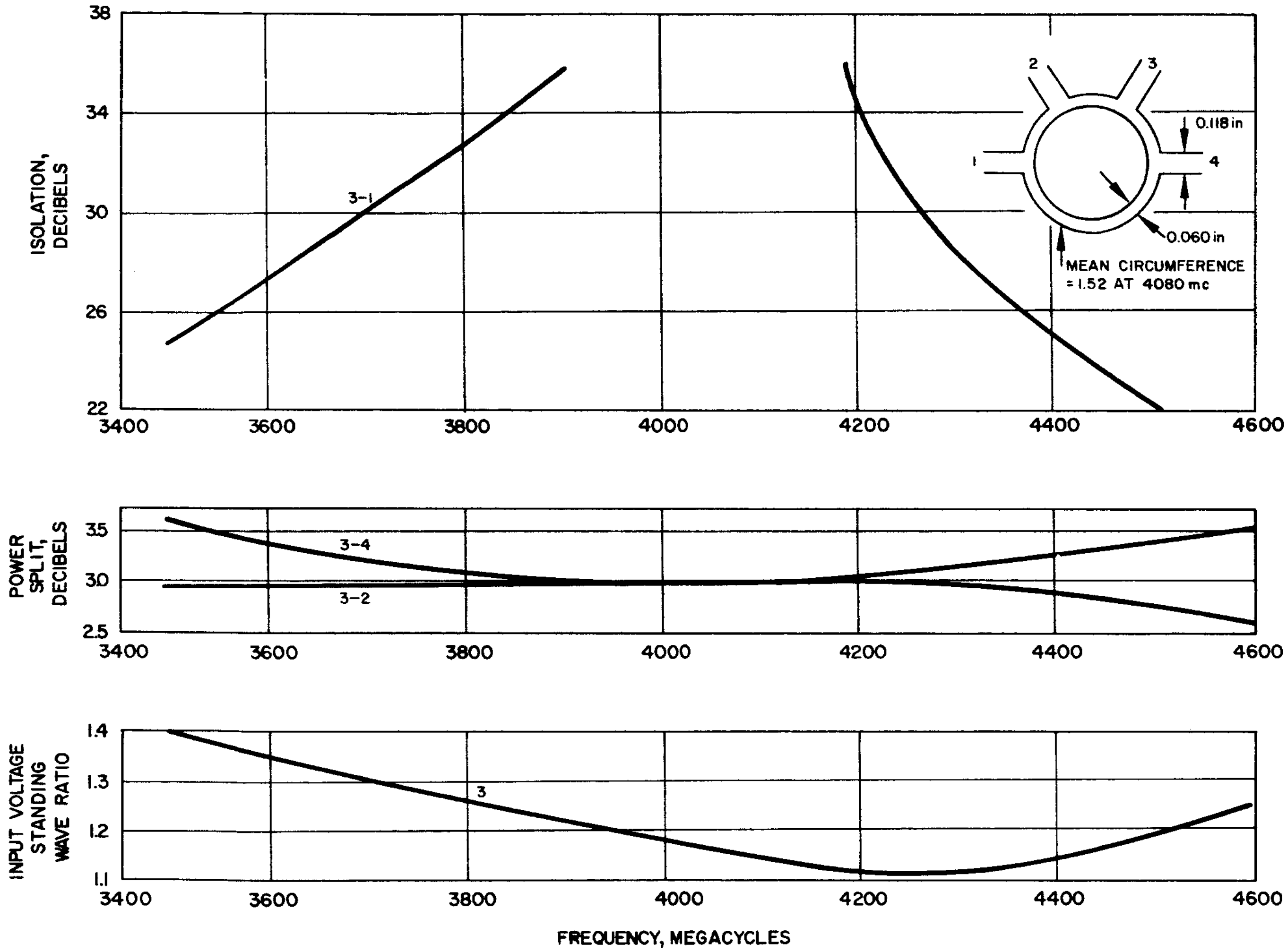


Figure 6-19. Characteristics of 4 kmc Hybrid Ring

In addition, research has shown that carbon film resistors can be used as reliable overload fuses. When overloaded, these resistors maintain their nominal value for a brief duration and then fail open (or to a very large resistance). This will allow the power system to be protected against short circuits in the power amplifiers which drive the ferrite phase shifter.

Jet Control Electronics and Solenoid Driver

The block diagram and all logic equations for the timing circuitry have been prepared. The logic is such that any quadrant command decoder can control any quadrant jet control timing circuitry.

Preliminary circuits with final configurations have been prepared for all timing circuits. The semiconductors to be used in these circuits have been selected and final passive component values will be prepared.

A solenoid driver, which uses a reasonable number of redundant components, has been constructed and is presently undergoing ambient temperature tests. Also, a reliability analysis is being conducted to determine if minor configuration changes could result in a more reliable circuit. The present configuration is undergoing final preparation.

A design inventory of the jet control electronics and solenoid drivers is being prepared.

CENTRAL TIMER

The present status of the Syncom II central timer is summarized below.

Milestones Met

1) Preferred core material selected (Orthonol).

2) Preliminary preparation of the timer selection switch has been made.

3) Preliminary preparation of the central timer block diagram has been accomplished.

4) Design of an incremental magnetic case pulse shaper and a count-of-10 scale stage has been built and the breadboard has successfully operated over the temperature range -35 to +70° C.

5) The preliminary test plan for the central timer has been prepared.

Problem Areas

1) The method of implementing static reset of apogee timer cores will depend upon the timing tolerance allowed for the apogee motor firing. If apogee motor timing of 315 minutes ± 1 percent is adequate, then a simple reset scheme can be incorporated. If tighter tolerances are required, then the reset must become more sophisticated. The above results from the requirement that the 2.81-minute output be available continuously so that it may be monitored on the gantry.

2) It appears quite undesirable to go to a variable timer. The added optimization of apogee engine firing time would have to be significant to compensate for the considerably reduced reliability that would occur.

Preliminary Specification

The dual function of the central timer in the Syncom II spacecraft is 1) to provide time-of-day correction signals to the phased array control electronics (these are provided at 2.81-minute intervals throughout the life of the vehicle); and 2) to generate the apogee motor ignition signal 315 minutes after separation from the second-stage booster. (See Figure 6-20.)

In conjunction with the Syncom II redundancy requirements, one timer is provided in each of the spacecraft's four quadrants. Due to the critical nature of the timer, additional redundancy is provided by the requirement in that any of the four timers are able to drive any PACE. Thus, the loss of a timer will not disable the 2.81-minute time-of-day correction inputs into any PACE.

Redundancy is also provided at the apogee motor driver input gate in that a minimum of two timers must trigger their 315-minute output latch before the apogee motor driver will be triggered. Thus, an early failure of any one timer will not erroneously ignite the apogee motor.

The inputs to the central timer are command +24 volts and command -24 volts. The timer outputs, for at least a temperature range from 0 to 50° C, are as follows:

1) Output at "t":

Type of output: pulse
Repetition rate: one pulse/2.8126 minutes
Repetition rate accuracy: ± 0.02 percent
Pulse width: 10 microseconds nominal
Levels:
    Quiescent: 0 volts
    Pulsed: 24 volts

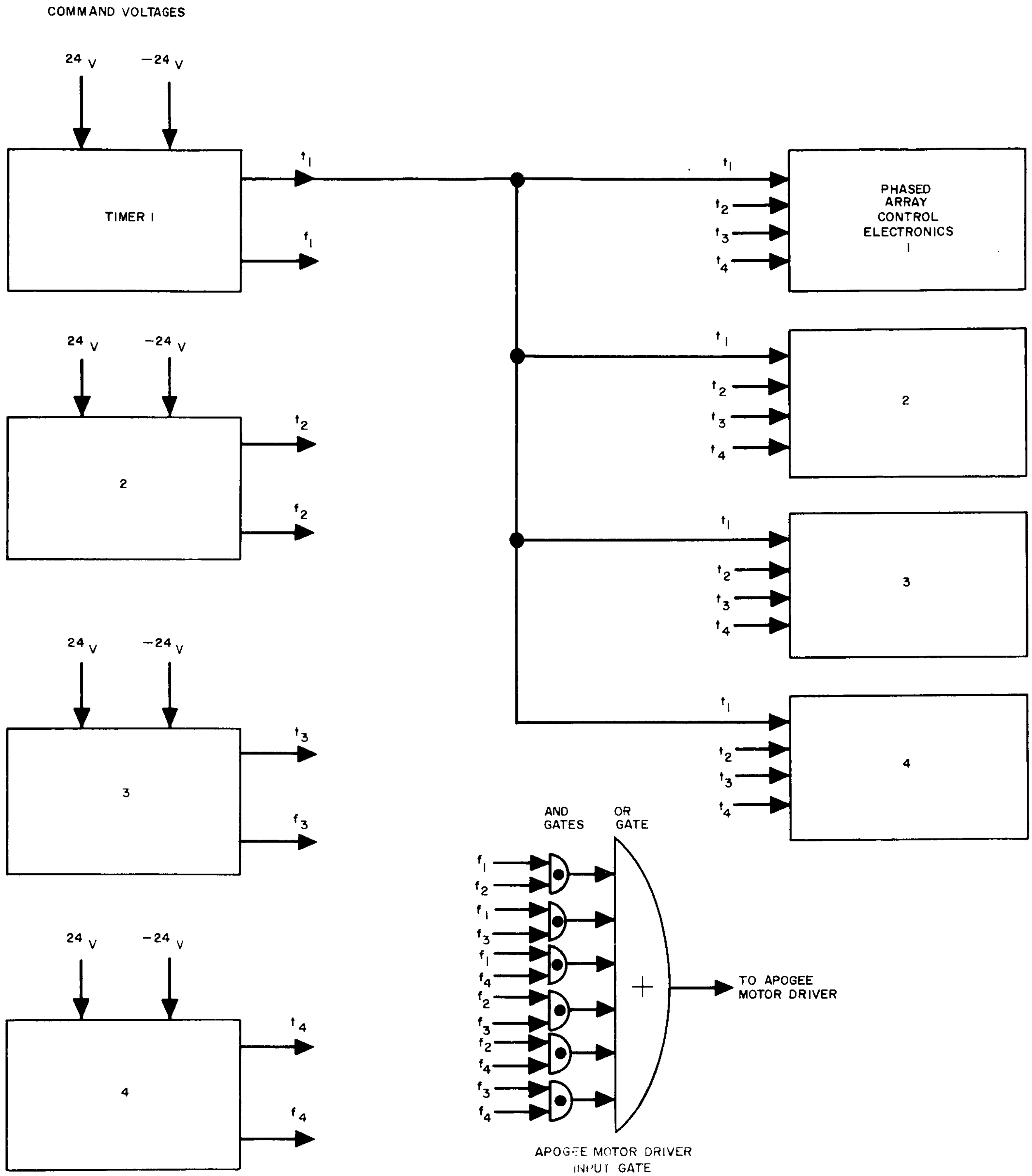


Figure 6-20. Syncom II Central Timer Block Diagram

2) Output at "f":

Type of output: level change from 0 to -24 volts
Timing: level change will occur 315.008 minutes after separation from second stage
Timing accuracy: +0.02 percent, -0.15 percent

COLLINEAR ARRAY RECEIVING ANTENNA

Cloverleaf Array (6 kmc)

Impedance data and array characteristics for the cloverleaf array were presented in the Summary Report. Additional patterns and gain measurements are presented here.

E-plane and H-plane antenna patterns have been made across the 200-mc frequency band. E-plane and H-plane patterns for the center frequency (6300 mc) and the ends of the frequency band (6200 and 6400 mc) are shown in Figure 6-21.

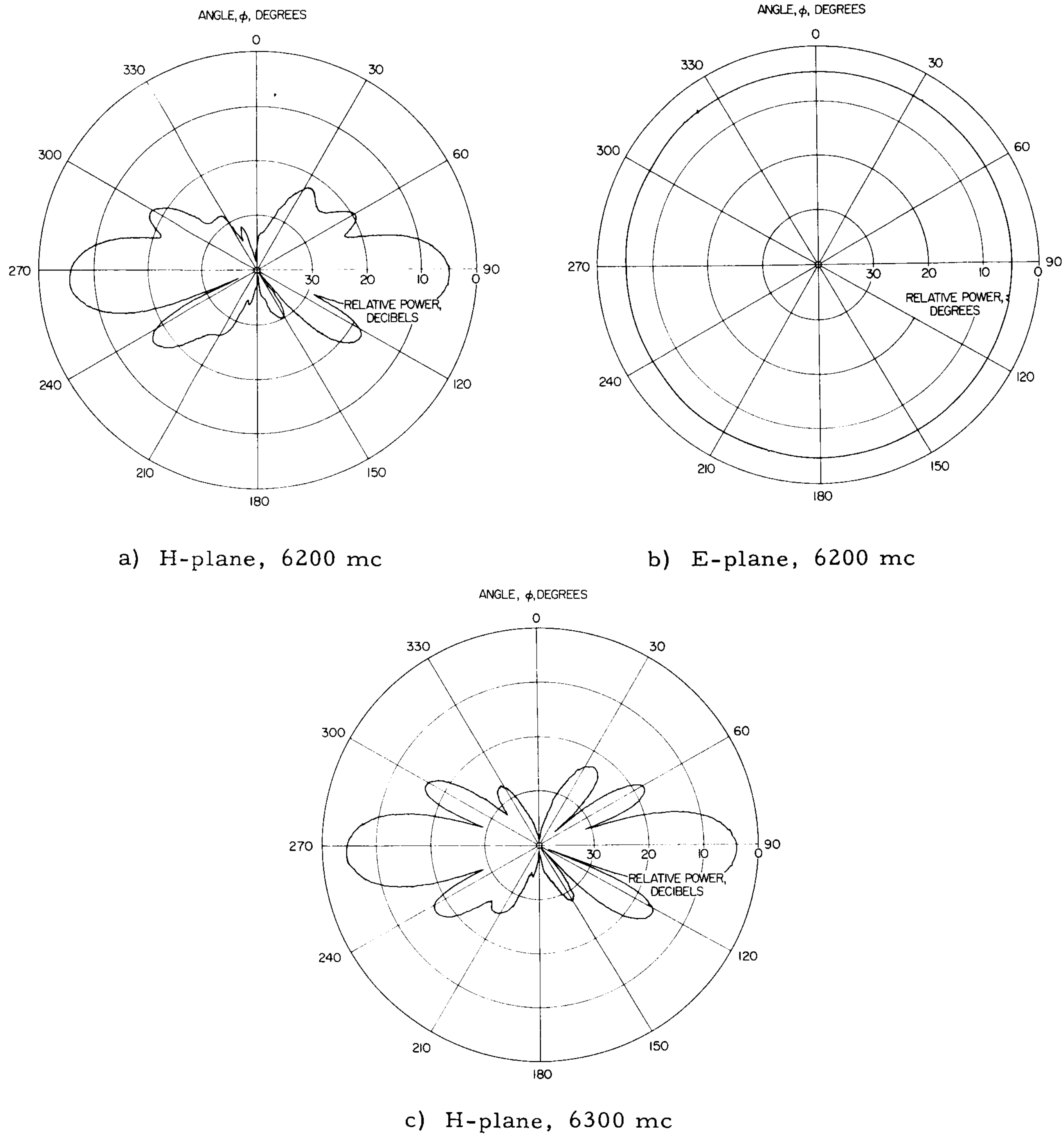
The gain of the antenna was measured across a 300-mc frequency band centered at 6300 mc. The results of this measurement are tabulated below:

Frequency, mc	Gain, db
6150	6.5
6200	6.6
6250	7.8
6300	8.1
6350	7.4
6400	7.0
6450	6.8

Biconical Dipole Array - Vertical Polarization

Testing is being continued on Model "3A"; electrical radiation pattern characteristics have been taken. The results in gain, beam width, and bandwidth were similar to the previous models and are well within the specifications.

Problems remain in the input impedance matching area, which, while being within a VSWR of less than 2:1, is not repeatable. The TM connector is being investigated accordingly.

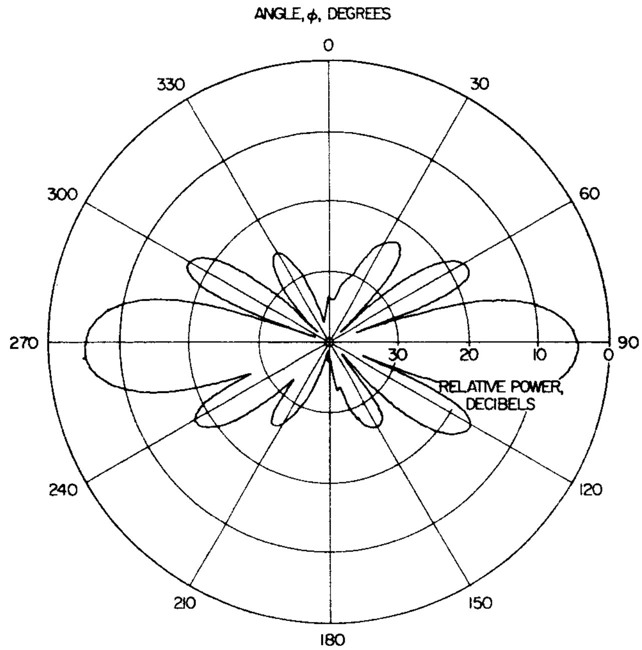


a) H-plane, 6200 mc

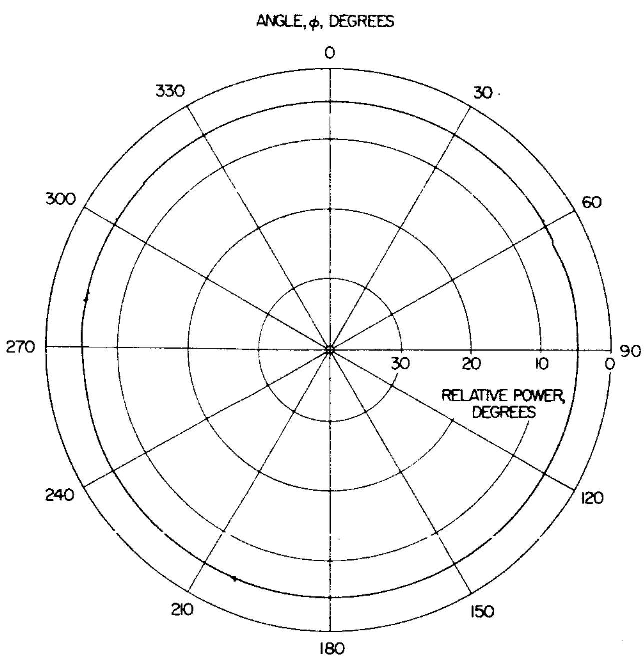
b) E-plane, 6200 mc

c) H-plane, 6300 mc

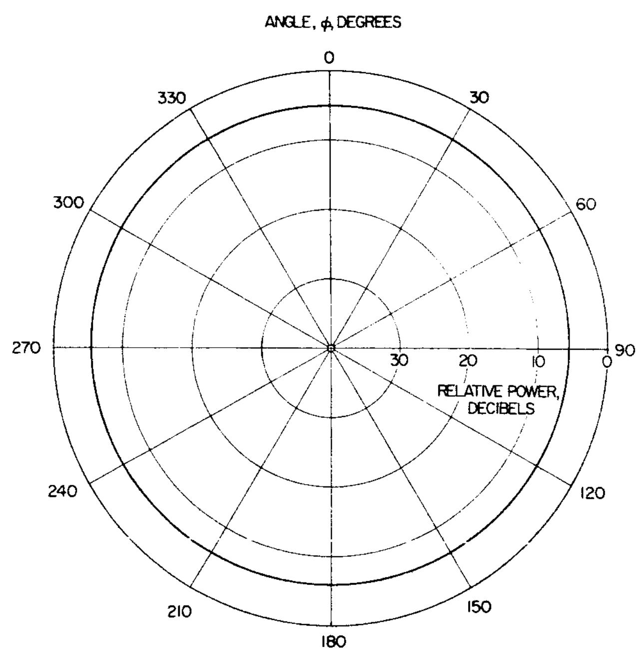
Figure 6-21. Six-Element Clover Leaf Array



d) E-plane, 6300 mc



e) H-plane, 6400 mc



f) E-planc, 6400 mc

Figure 6-21 (continued). Six-Element Clover Leaf Array

Modes of Spacecraft Orientation Maneuver Program for Computer Analysis of Stability

Dynamics Analysis of Spacecraft Orientation Maneuver

A program to study the dynamics of the orientation maneuver has been generated for the IBM 7090. Equations 6-1 through 6-8 are the basic equations. Coordinate transformation equations are omitted for brevity. The various axis systems used are: 1) inertially-fixed centered in spacecraft cg, 2) body-fixed in spacecraft cg, 3) spacecraft body-fixed centered at nozzle at hinge line, and 4) spacecraft body-fixed centered at nozzle at hinge line but rotated with y axis on hinge line (Figure 6-22).

The basic equations of motion for the spacecraft in body-fixed coordinates are

$$\dot{p} = \frac{1}{A}\left[(B - C)\, q \cdot r + L_T \cdot \frac{W_o}{W}\right]$$

$$\dot{q} = \frac{1}{B}\left[(C - A)\, p \cdot r + (M_T + M_D)\frac{W_o}{W}\right] \qquad (6\text{-}1)$$

$$\dot{r} = \frac{1}{C}\left[(A - B)\, p \cdot q + (N_T + N_D)\frac{W_o}{W}\right]$$

$$W = W_o + \dot{W}_t = W_o - (F^t/32.2I)$$

Equations for the Eulerian angles are

$$\dot{\theta} = q\cos\phi - r\sin\phi$$

$$\dot{\psi} = (q\sin\phi + r\cos\phi)/\cos\theta \qquad (6\text{-}2)$$

$$\dot{\phi} = p + \dot{\psi}\sin\theta$$

for order of rotation of roll, pitch, and yaw.

L_T, M_T, and N_T are defined in the following equations and are the moments imposed by the jets about the respective axes of the spacecraft.

$$\left\{\begin{aligned} X_T &= -F\left[\cos\delta_T\cos(\Delta\delta_T) - \sin\delta_T\sin(\Delta\delta_T)\right] \\ Y_T &= -F\left[\sin\delta_T\cos(\Delta\delta_T) + \cos\delta_T\sin(\Delta\delta_T)\right]\sin\alpha_T \\ Z_T &= +F\left[\sin\delta_T\cos(\Delta\delta_T) + \cos\delta_T\sin(\Delta\delta_T)\right]\cos\alpha_T \end{aligned}\right. \qquad (6\text{-}3)$$

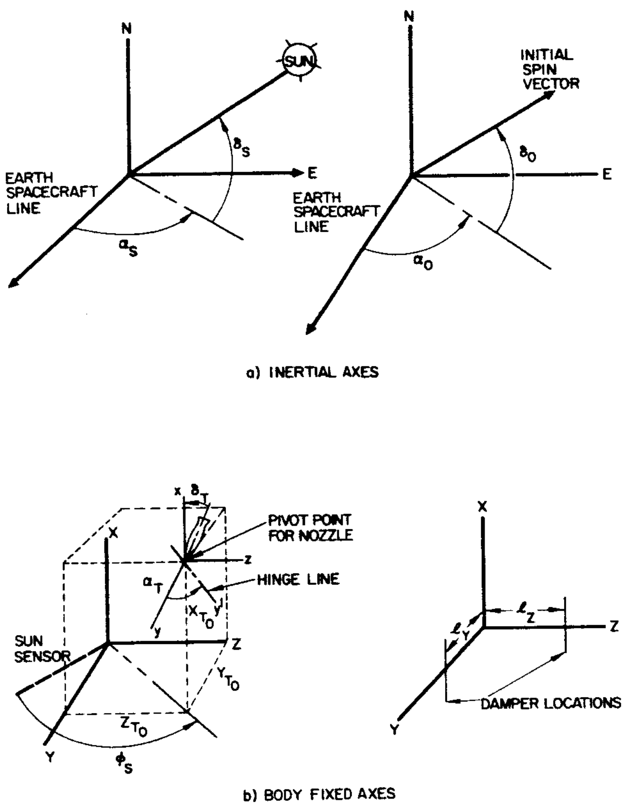


Figure 6-22. Coordinate Systems for Syncom II

$$
\begin{cases}
x_T = x_{T0} \\
y_T = y_{T0} + \eta \cos\alpha_T - \zeta\left[\cos \delta_T \cos(\Delta\delta_T) - \sin\delta_T \sin(\Delta\delta_T)\right]\sin\alpha_T \\
z_T = z_{T0} + \eta \sin\alpha_T + \zeta \cos \delta_T \cos(\Delta\delta_T) - \sin\delta_T \sin(\Delta\delta_T)\cos\alpha_T
\end{cases}
\tag{6-4}
$$

inputs

$$
\begin{cases}
L_T = -(z_T)(Y_T) + (y_T)(Z_T) \\
M_T = -(x_T)(Z_T) + (z_T)(X_T) \\
N_T = -(y_T)(X_T) + (x_T)(Y_T)
\end{cases}
\tag{6-5}
$$

The W_o/W terms multiplying the L_T, M_T, N_T, M_D, N_D terms in Equation 6-1 are the result of the approximation that A, B, and C are changed in only a minor way by the full mass loss, and are accounted for in the other terms. This saves computer time and programming.

M_D and N_D denote the moments put in by the nutation dampers. These equations were determined by scaling some experimental data from tests on the nutation dampers of Syncom I.

$$
\begin{aligned}
\mathring{\xi}_y &= -\left[K_y\left(\frac{A-B}{C}\right)p\right]\xi_y + \ell_y(\mathring{r} - p\cdot q) \\
\mathring{\xi}_z &= -\left[K_z\left(\frac{A-C}{B}\right)p\right]\xi_z + \ell_z(-\mathring{q} - pr)
\end{aligned}
\tag{6-6}
$$

$$
\begin{aligned}
M_D &= W_{Dy}\left[K_y\left(\frac{A-B}{C}\right)p\right]\xi_y\,\ell_z \\
N_D &= W_{Dy}\left[K_z\left(\frac{A-C}{B}\right)p\right]\xi_z
\end{aligned}
\tag{6-7}
$$

where ξ_y and $\mathring{\xi}_y$ are the velocity and acceleration of the cg of damper fluid, respectively.

δ_T is defined by its equation of motion in its plane of motion as follows:

$$\overset{\circ\circ}{\delta}_T = -\overset{\circ}{q}_T - \left(\frac{\rho^2_{Ki}}{\rho^2_K}\right)\left[(p_T^2 - r_T^2)\sin\delta_T\cos\delta_T + p_T r_T(\cos^2\delta_T - \sin^2\delta_T)\right]$$

$$+\frac{\bar{\rho}_T}{\rho_K^2}\left\{\left[p_T(p_T x_{TT} + q_T y_{TT} + y_T z_{TT}) - \dot{r}_T y_{TT} + \dot{q}_T z_{TT} - x_{TT}(p_T^2 + q_T^2 + r_T^2)\right](\sin\delta_T)\right.$$

$$\left. + \left[r_T(p_T x_{TT} + q_T y_{TT} + y_T z_{TT}) - \overset{\circ}{q}_T x_{TT} + \dot{p}_T y_{TT} - z_{TT}(p_T^2 + q_T^2 + r_T^2)\right](\cos\delta_T)\right\} \qquad (6\text{-}8)$$

$$-\frac{K_T}{W_T\rho_K^2}(\delta_T - \bar{\delta}_T) - \frac{J_T}{W_T\rho_K^2}(\overset{\circ}{\delta}_T)$$

The subscripts T or TT denote quantities transformed to a coordinate system fixed in the hinge line (y axis) with origin at the nozzle attach point where F is the thrust of the nozzle approximated by a trapezoid shown in Figure 6- 23.

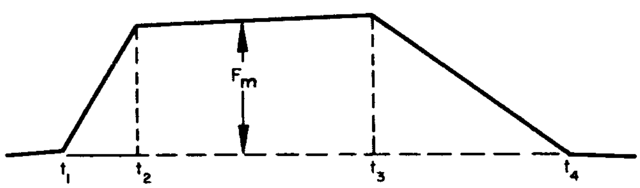


Figure 6-23. Pulse Definition

Some of the important items that can be easily changed in the program are:

1) Initial angles and rates, both inertial and body-fixed
2) Firing angle from sunline
3) Sunline and spin vector orientation
4) Jet pulse dimensions
5) Geometry of nozzle and hinge line location
6) Inertial properties of spacecraft and nozzle.

The only assumption made is that the motion of the jet does not affect the motion of the satellite. This is reasonable because the mass and moments of inertia are two orders of magnitude below those of the basic satellite.

The integration scheme is such as to automatically decrease the integration interval during pulsing while using an optimum time interval during the rest of the cycle to conserve machine time. Print-out intervals during pulsing are also decreased to study detailed effects of pulsing.

The pulsing is triggered by passing the sunline. Nutation dampers can be be simulated in each axis, even though there is only one in the spacecraft to allow for contingencies. Printouts include inertial as well as body-fixed quantities.

During the initial phases of program checkout, it was estimated that excessive running time would be incurred because of the high frequency of the nozzle motion coupled with the automatic integration interval feature. It was therefore decided to use a fixed nozzle. (Fixed-nozzle misalignments can be studied, however.)

All subroutines and features are now code checked and the first trial run of the total program is being made.

Nomenclature

W_o = initial spacecraft mass, slugs

W_T = nozzle mass, slugs

A = spin axis moment of inertia, slug ft^2

B = pitch axis moment of inertia, slug ft^2

C = yaw axis moment of inertia, slug ft^2

x_{T0}, y_{T0}, z_{T0} = coordinates of nozzle at hinge line, feet

α_T = angle defining hinge line, degrees

$\dot{\delta}_{T,0}$ = initial deflection rate of nozzle, rad/sec

$\bar{\rho}_T$ = distance from hinge line to nozzle, feet

ρ_K = nozzle radius of gyration about hinge line, feet

ρ_{K1}^2 = difference of squares of radius of gyrations of nozzle lateral and thrust axes, $feet^2$

K_T = nozzle spring constant, ft lb/rad

J_T = nozzle damper constant, ft lb sec/rad

F_M = peak nozzle thrust, pounds

I = nozzle propellant impulse, seconds

η = thrust offset along nozzle y axis, feet

ζ = thrust offset along nozzle z axis, feet

$\Delta\delta_T$ = angular nozzle thrust misalignments, degrees

K_y = damper constant

ℓ_y = damper moment arm, feet

W_{Dy} = mass of damper fluid, slugs

K_z = damper constant

ℓ_z = damper moment arm, feet

W_{Dz} = mass of damper fluid, slugs

p, q, r = angular rates about body-fixed axes, rad/sec

$\bar{\delta}$ = final spin vector orientation

α_s = sunline orientation, degrees

δ_s = sunline orientation, degrees

α_o = initial spin vector orientation, degrees

δ_o = initial spin vector orientation, degrees

ϕ_s = firing angle to center of pulse, degrees

Spin Rate Mechanism Test Data

Testing during the report period proceeded according to the general objectives stated in the Summary Report, namely, to provide operational data on a specific mechanism embodying the basic spin rate control design features (torsion spring propellant lines, flexure pivots, and viscous damper). Test results have been obtained in the Summary Report outline areas:

1) Static test

2) Transient decay test

3) Centrifuge test

4) Vibration test

The centrifuge test is complete. However, further testing of bellows dampers, both transient decay and vibration, will be required to:

1) Determine the practical limitations on increased damping with the bellows damper.

2) Determine the feasibility of operating an otherwise satisfactory damper design in a vibration field without a pin puller.

Static Test

Spring rate measurement: Spring rates of the individual spring rate-contributing elements (flexure pivots, torsion coils, and bellows damper) were determined on the bench to provide orders of magnitude for future design estimates and to provide basic data for check calculations during the current test series.

Within experimental error, the spring rates of the three elements were constants, having the values:

Flexure pivots	0.175 in-lb/deg (Figure 6-24a)
Torsion coils	0.653 in-lb/deg (Figure 6-24b)
Bellows damper	0.724 in-lb/deg (Figure 6-24c)

Nulling error: With the torsion coils set for zero bias torque at zero degrees deflection of the jet, the tendency of the unit to hang off of null was measured. The spin rate control unit was deflected ± 8 degrees from the null and allowed to return slowly toward null. The resultant error was:

For all three spring elements active - $< 1/4$ degree.

For damper and flexures only - not apparent.

For coils and flexures only - $< 1/4$ degree (about the same as the first case).

For a 4 to 5 degree initial condition, no apparent error was noted.

Conclusion: Nulling errors due to the bellows damper and flexure pivots are negligible.

Transient Decay Test. The aim of the transient decay tests in this period was to check the variation of the damper performance with simulated temperature environment. With 750 centistoke silicone fluid assumed as nominal, due to its relatively satisfactory performance (decay time of $\sim$ 0.30 second versus $\sim$ 0.46 second per revolution of the spacecraft-maximum), and after consulting viscosity-temperature data, 2000 centistoke silicone fluid was chosen as representative of 0° F and 450 centistoke silicone fluid as representative of 135° F.

Representative transients of the model spin rate control for 750 cs and 2000 cs oil are plotted in Figures 6-24d and 6-24e. Although the actual transients do vary in shape, the effective damping times are close ($\sim$0.25 versus $\sim$0.30 second).

From the above data it can be reasonably concluded that, for the temperatures and range of viscosities specified, little variation in damper performance can be expected.

In the course of the tests it was decided not to run the lighter oil and direct the investigation more toward obtaining appreciably greater damping on the presumption that a damper not too overdamped, but more highly damped than at present, would enhance feasibility of the unit's vibration operation without a pin puller. Experimentation with both fluids of higher viscosity and damping orifices of smaller size are proposed for a continuation of damper development.

Centrifuge Test. The objective of the centrifuge test was to check the nominal operation of a spin rate control model embodying the basic design features noted above. To establish such operation, the spin rate control model was mounted on the arm of a 32-inch centrifuge with the pivot axis at 45 ± 2 degrees to the arm centerline. Since it had already been established that the bellows damper acts like a linear spring for design travel and the most convenient location for the shaevitz position transducer was at the damper location, the centrifuge test was conducted without the damper (from a structural standpoint the vibration testing proposed presents a more severe structural environment for the damper than the centrifuge test). Figure 6-24f shows the centrifuge setup (not shown is a plexiglass windshield which covered the unit during test).

The test was conducted in two parts (after calibration of the position transducer): 1) A series of speeds giving operation over a ± 8-degree range and 2) an overspeed test. The results of the speed versus angle test are plotted in Figure 6-24g. (The plotted speeds have been converted from the test arm radius to the spacecraft mounting radius, noting the tradeoff between radius and speed squared.)

Based on this data, the following is objective:

1) The action of the control is continuous.

2) The relationship between speed and angle is the expected ($F = m\omega^2 r$, $T = Fd$, $T = K\theta$) square law.

3) After setting the torsional coils to a theoretically predicted 26 in-lb on a separate jig and then installing the coils in the spin rate control model the zero degree (control deflection) speed is only 1 percent in error, showing the feasibility of the approach from this standpoint. (The bellows and flexures will not appreciably affect this point since they both are nominally null at zero degrees.)

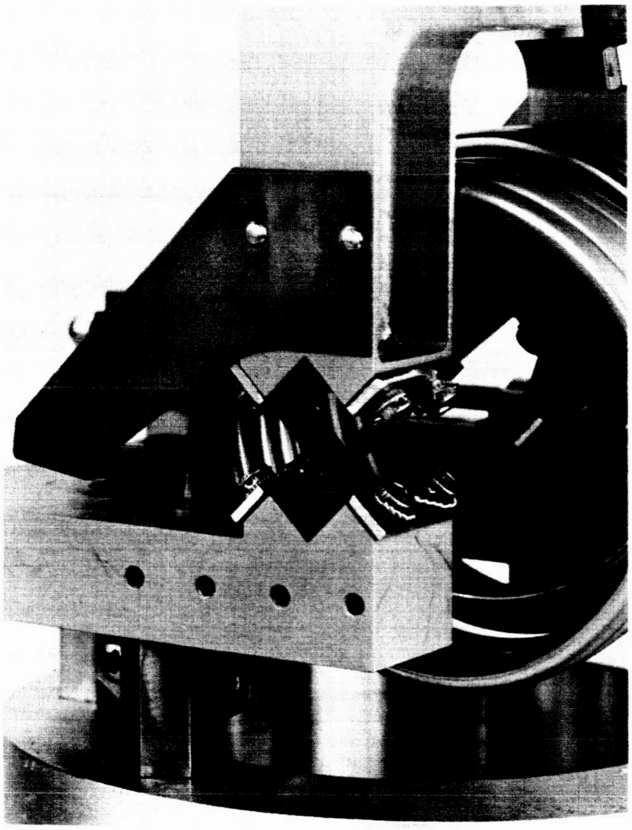
4) In the design speed range, the speed-angle relationship is almost linear.

5) The range of the control, as tested, is less than the design range of 75 to 125 rpm. (This is due to the net spring rate of coils plus flexures being less than the design spring rate due to the absence of the bellows. Simple extrapolation of above results, allowing for the bellows spring rate, shows the design range is readily obtained by spring adjustment.)

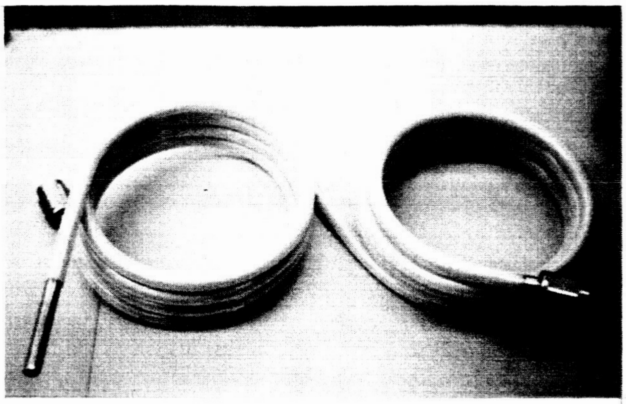
The unit was oversped to A ~ 140 rpm equivalent at 26 inches radius, or ~ 14 g, compared to 11 g at 125 rpm. No apparent damage was sustained. The Teflon covering of the torsion coils, Figure 6-24b (for protecting the coils on subsequent vibration tests), had no appreciable effect on centrifuge operation.

Vibration Test. The primary purpose of the vibration testing is to observe the performance of the bellows damper in vibration fields as high as 50 g in restricted frequency bands and 15 g over a wider range. A secondary purpose is observation of the coils and flexures under the same conditions. Two test configurations are proposed, with and without pinpuller.

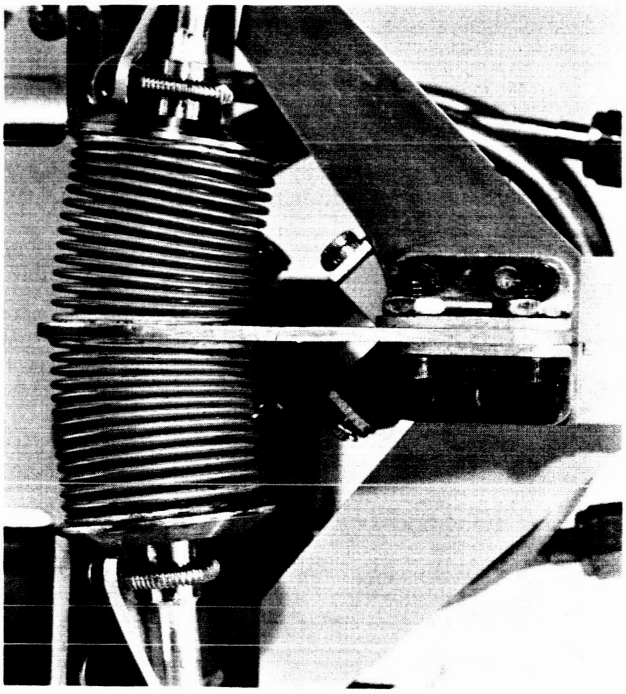
Initial vibration testing has been performed at low levels with the pinpuller installed with no visible damage. Levels of 4 and 6 g (or 1/2-inch double amplitude) over a frequency range of 5 to 2000 cps were chosen, with separate 10 minute sweeps for all axes. All tests were



a) Flexure pivots

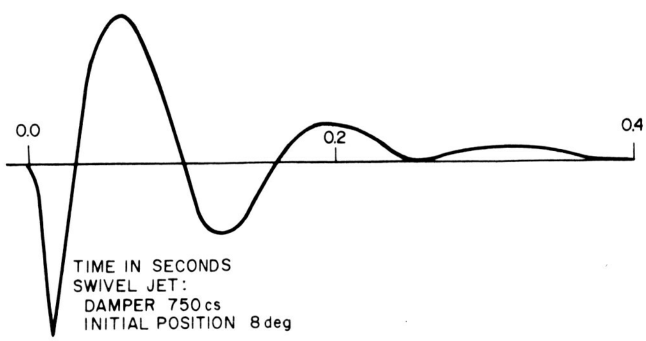


b) Torsion coils

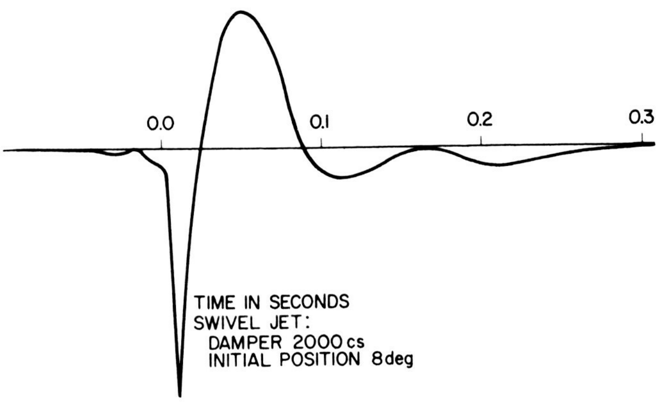


c) Bellows damper

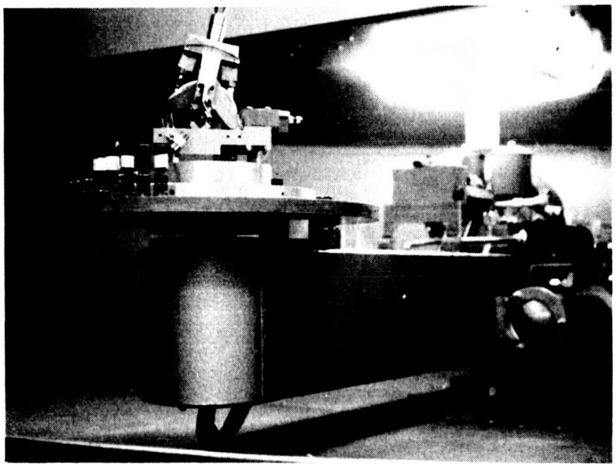
Figure 6-24. Syncom II Tests



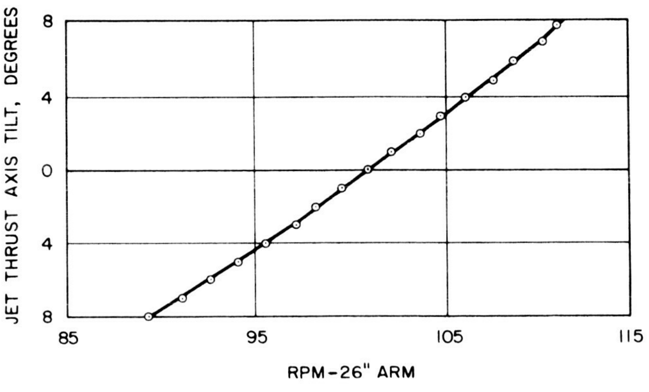
d) Transient decay of spin rate control for 750 centistoke silicone fluid



e) Transient decay of spin rate control for 2000 centistoke silicone fluid



f) Centrifuge



g) Spin rate control centrifuge

Figure 6-24 (continued). Syncom II Tests observed visually with the aid of a slip-sync light. No outstanding motions of the bellows or flexures were noted. However, the coils showed resonance effects between 150 and 170 cps.

During the next report period higher vibration levels both with and without the pinpuller will be performed.

Miscellaneous Tests. During the next report period an endurance test on the bellows damper design will be performed, in addition to the qualification vibration testing on the spacecraft. Review of the latter test results may allow modification of the spin rate control vibration specification.

Bipropellant Reaction Control System Development

Development of the reaction control system has been subcontracted to the Marquardt Corporation. Details of the development program during this report period are covered in Marquardt Monthly Progress Report MR-1-2, Appendix C of this report. A summary of the progress to date follows.

On 17 April 1963, Marquardt had expended the funds allotted for Phase I development. The program was temporarily placed in abeyance. Technical difficulties experienced in manufacturing and testing thrust chambers have resulted in a program negative slack of 9 weeks which includes a 2-week contingency period. Completion of Phase I is now scheduled for 15 July 1963.

A series of sea level and altitude test firings was conducted with both stainless steel and molybdenum chambers. The objectives of these tests were to isolate instrumentation difficulties, evaluate methods for their resolution, and generate engine performance data with increased reliability.

Three molybdenum chambers failed during the reporting period. The first two failures were attributed to manufacturing defects and improved methods of nozzle fabrication were instituted. A third failure resulted with an engine fabricated by improved techniques. During the interim, flow meter calibration and measurement procedures were reviewed. Based on the findings of a subsequent flow meter calibration development program (now of a continuing nature) there were significant improvements in test instrumentation evolved. In addition, a technique for attaching thermocouples to the molybdenum disilicide coated chamber was developed. With these improvements incorporated, a recent test firing of a molybdenum chamber resulted in conclusive evidence that the chamber wall temperature in the vicinity of the throat at optimum oxidizer-to-fuel ratio was exceeding the limits (3000° F) of the disilicide coating. Hughes and TMC are currently evaluating alternate solutions. The most expedient solution to the overheating problem would be to run the axial engines on a pulsed duty cycle. Preliminary discussions indicate that this is acceptable from a trajectory correction standpoint. Early testing at Marquardt shows that a steady state temperature of 2500° F will be reached with a 33 percent duty cycle.

A Syncom II spacecraft center structure was received by TMC. Fabrication and installation of the engineering model propulsion unit is 60 percent complete.

Fabrication and assembly of the breadboard propulsion unit, exclusive of the engine and swivel mechanism assemblies, were completed. Design of the swivel mechanism was completed and subcontracted for fabrication. The Marquardt altitude test sphere in which a feasibility demonstration with the breadboard model will be run was completed. Spin table and breadboard model installation was completed and the facility was held at the required test pressure of 500 microns for 10 minutes.

TELEMETRY AND COMMAND

Telemetry and Command Antenna Design

An RF mockup of the spacecraft constructed of a lightweight skin is being fabricated for preliminary antenna studies. This mockup will be used for both impedance and preliminary radiation pattern measurements.

A two-antenna system appears to be the most feasible method of operation for the four telemetry transmitter, four command receiver system of the Advanced Syncom spacecraft. Interaction studies, matching, and hybrid-balun design will be experimentally verified for this approach. The Syncom I design can be utilized for each system. Because of the difference in spacecraft size, however, new layouts will be investigated. A lower loss, but somewhat heavier, coaxial is being considered for forming the hybrid balun. Tradeoffs between reduced loss and added weight will be studied.

Telemetry Encoder Requirements

The Syncom II telemetry encoder is essentially the GSFC-PFM system with the following parameters.

Channel rate:	48.5 $\pm$0.5 percent channels/sec
Subcarrier oscillator:	10 kcps $\pm$50 percent
Reference burst:	Greater than 15.6 kcps, noncoherent
Synchronization frequency:	4500 cps
Data accuracy:	Encoder error less than 1.0 percent
Number of frames:	To be determined

Digital information, including command register contents, is encoded four bits per channel for a 16-level system according to the requirements listed in Table II of the GSFC-PFM telemetry standards.

Command Decoder

A detailed block diagram of the dual-mode command decoder is complete and the detailed system description is being prepared. The operation of the decoder is, with a few modifications, essentially the same as that described in the Syncom II Summary Report.

A system design review is planned, during which the operation of the dual-mode decoder will be compared to two single-mode systems, the frequency shift keyed type and the single tone interrupt type. These two single-mode types are combined in the dual-mode system.

ELECTRICAL POWER

Power System Summary

The solar panels shall be capable of supplying total power to the spacecraft electronics system during the five-year orbital period. The batteries shall be capable of handling the spacecraft electrical requirements during launch, ecliptic and orientation periods. There shall be two power systems as shown in Figure 6-25.

System Design

The power system shall operate in the following manner:

Launch. During launch the solar cells will be dark causing the load to swing the solar bus voltage toward zero; as the voltage passes a pre-determined voltage level the logic-inhibit circuit will fire the SCR (Figure 6-26) allowing the battery to power the load.

Orbital Operation. Solar energy shall be converted to electrical power by solar cells to drive the load and charge the batteries. When the batteries require charging, the third electrode in each cell sends a signal to the logic-inhibit circuit. If the battery is not being used by the load, the boost-add charge circuit will be activated recharging the batteries.

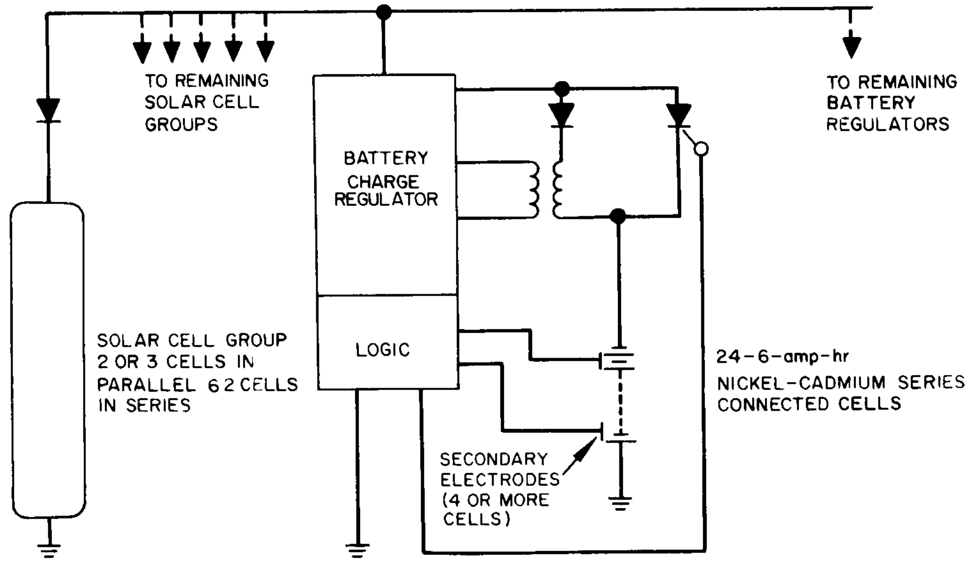


Figure 6-25. Electrical Power System Schematic

Power System Design Specifications

1.0 SPACE ENVIRONMENTAL CONDITION

1.1 General: The SYNCOM II power system is designated to furnish sufficient electrical power to operate the spacecraft electronics equipment during a five-year period in a stationary synchronous orbit. The battery power will be supplied for a three-week period twice a year. The maximum time of discharge for each period shall be 1.15 hours each 24 hours. (See Figure 6-27.

1.2 Variations: Variations in solar intensity of ± 3 percent to the mean have been included.

1.3 Cell Power Output: Solar cell output power is based on variations in panel temperatures as follows:

a) Normal incidence $\beta = 0°$	$75° \begin{smallmatrix} +5° \\ -10° \end{smallmatrix} F$
b) Oblique incidence $\beta = 25°$	$60° \begin{smallmatrix} +5° \\ -10° \end{smallmatrix} F$
c) Lowest temperature during maximum eclipse time	-155°F

1.4 Battery Power: Battery power is based on variations in temperature between 40°F and 100°F.

1.5 Radiation Damage: Radiation damage is considered from two sources: a) solar flare activity, and b) Van Allen belts. Precautions have been taken to reduce damage to a minimum.

1.6 Micrometeorite Damage: Micrometeorite damage will be slight.

2.0 ARRAY POWER OUTPUT

SYNCOM II power system will supply the necessary energy to operate the spacecraft electrical equipment. Figure 6-28 represents the calculated array performance.

3.0 ARRAY REQUIREMENTS

3.1 The solar array components listed below are designed to operate for the full five-year period under space conditions set forth in Section 1.0.

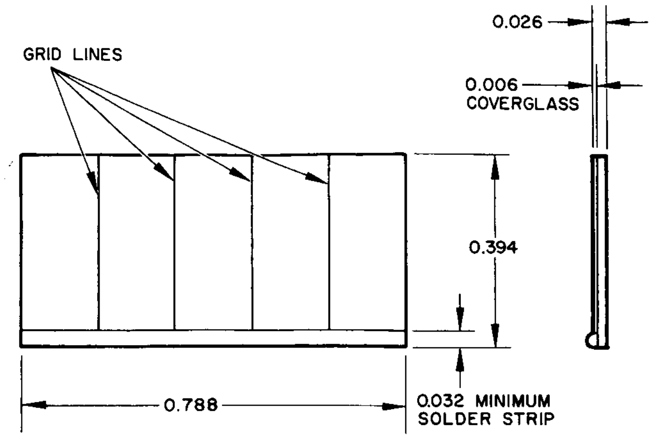


Figure 6-26. Silicon Solar Cell

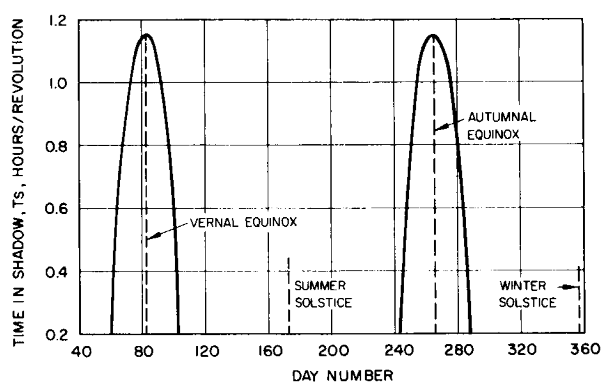


Figure 6-27. Shadow Time for Satellite in 24-hour Equitorial Orbit

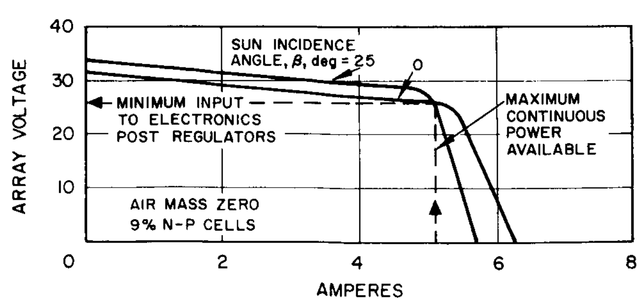


Figure 6-28. Solar Array Characteristics

3.1.0 Solar Cells: The solar cell shall be a silicon "N on P" junction type, 1 by 2 centimeter size, with coverglass applied. The individual solar cells shall be covered with a coverglass capable of meeting the requirements specified below. The solar cells will be capable of being bonded to a honeycomb substrate, and will be capable of being electrically interconnected in series-parallel groups to assure maximum reliability of operation.

3.1.1 Coverglass: Each solar cell shall be covered with a 0.006-inch thick Corning number 0211 microsheet glass coated with an anti-reflection coating on the top surface and an ultraviolet reflecting coating on the under surface.

3.1.2 Absorptance and Emittance: Average emittance of the solar cell top surface with the coverglass applied shall not be less than 0.83 from 25°C to 125°C. Average absorptance to solar radiation in the wavelength region 0.2 to 2.5 microns shall not exceed 0.82.

3.1.3 Spectral Transmittance: The spectral transmittance of the cell cover shall meet the following requirements:

	Wavelength (Microns)	Transmittance Percent
	0.400 ± 0.015	50
From	0.300 to 0.370	Less than 1 average
From	0.500 to 1.000	92 minimum

3.1.4 Negative Contact: The exposed negative (top) contact of the cell shall have a clean, uniform and complete line of solder along the 2 cm dimension. The width of this solder line shall not be less than 0.032 inches.

3.1.5 Positive Contact: The positive contact of the cell shall be flat within 0.005 inch and free of all contaminating material. The surface shall be optimized so as to provide maximum adhesive bonding strength.

3.1.6 Weight: The total assembled solar cell weight including coverglass shall not exceed 0.35 grams average per lot.

3.1.7 Power Output: The power output of each solar cell with coverglass applied under mass zero spectral conditions, and solar radiation intensity of 140 mw/cm^2 shall meet the following requirements:

Test Conditions

Temperature °C	Voltage Volts	Power Milliwatts
25 ± 2	0.46 ± 0.010	22.7 minimum

The specified voltage shall be used in measuring the power output of the solar cells.

3.1.8 Radiation Resistance: The cell shall comply with NASA-GSFC Specification No. 63-106 dated October 1962.

3.2.0 Solar Array

3.2.1 Power Output: The power output of the solar panels is based on zero air mass, temperature of 77°F, and solar intensity of 140 mw/cm^2 and may be summarized as follows:

Total cells per string	62
Total cells per group	186
Total groups per panel	8
Total cells per panel	1488 (on 14 panels) 1364 (on 2 panels)
Total panels per spacecraft	16
Total cells per spacecraft	23,560
Sun incidence angle, β	25 degrees
Effective groups illuminated, $0.636 \times 64 \cos \beta$	36.83
Effective area of illuminated cylinder	0.636
Power output per group at normal incidence $186 \times 22.7 \times 10^{-3}$	4.22 watts
Total power output with 8% degradation, $\beta = 25$ degrees	143 watts
Output voltage 0.46×62	28.5 volts
Blocking diode voltage	0.7 volt
Net array voltage	27.8 volts
Total array current	5.14 amperes

3.2.2 Panel Substrate: The solar panels consist of an aluminum honeycomb core set between fiberglass sheets. The sheet fibers run parallel to the panel edges. Fiberglass sheets are bonded to the core with an epoxy resin. Epoxy syntactic foam is used to fill in the edges.

3.2.3 Cell Mounting: Panel-cell mounting shall be arranged to maximize efficiency and minimize weight. Solar cells are mounted to the fiberglass-faced, aluminum honeycomb panel with an epoxy resin capable of holding the solar cells in position throughout the required life of the spacecraft.

3.2.4 Cell Configuration: The cell configuration shall be arranged for maximum flexibility. The solar array is composed of flat-mounted N on P solar cells on fiberglass-faced, aluminum honeycomb substrate panels. Sixteen panels are used to form the outer cylindrical surface of the spacecraft. Parallel interconnections at the cell level enhance solar array reliability. The physical wiring of the individual cell groups has been segregated by connecting alternate cell groups to provide maximum system flexibility. This interconnection method allows two separate and isolated solar array outputs to be utilized. Changes from one to two distinct buses can be accomplished by only a spacecraft harness change with no redesign or rework to the solar array. See Figure 6-25.

4.0 RADIATION ENVIRONMENT

Components shall be selected which will remain functional for five years in the radiation areas 22,500 miles from the Earth's surface.

4.1 Sources of Radiation: Two radiation sources may exist in the SYNCOM II orbit which could result in reduction of the solar array output: solar flare activity and Van Allen radiation. With the 6-mil glass covers similar to those used on SYNCOM I, the Van Allen radiation will cause less than 0.2 percent per year solar cell output degradation. Solar flare activity will result in a predicted 8 percent degradation in the last two years of the predicted five-year functional life of the spacecraft.

Micrometeorites are not considered to degrade the basic array output since a typical collision will result only in a small non-shorting cell puncture, which will be so small compared to the total area that the overall effect will be negligible.

5.0 BATTERY SYSTEM DESIGN

Battery system shall be adequate for supplying spacecraft power during boost and eclipse periods, also energy for pulse loads such as control system valve solenoids and the apogee motor igniter.

5.1 System: The battery system consists of four separate nickel-cadmium batteries of 24 cells each. The cells, rated at 6 ampere-hours each, are hermetically sealed and use sintered plate construction. Four or more of the 24 cells in each battery will have a sensory electrode whose output is proportional to the state-of-charge of the cell. The sensory electrode current is proportional to the cell state-of-charge and is used to terminate battery charging. The cell utilizes flat-plate construction with both the input and output terminals electrically insulated from the cell case. Electrical insulation of the terminals from the case allows direct thermal conduction to the spacecraft structure resulting in more uniform battery cell temperatures, hence potentially increased system reliability.

5.2 Battery Cell Requirements: Battery design shall be capable of providing power throughout the launch and orbital eclipse periods. The design of the battery depends on the bus voltage desired, dark-time power load, depth of discharge, solar array recharging rate, time available for recharge, and reliability requirements. Twenty-two series-connected cells would be required to furnish the minimum voltage (26 volts) to the electronics subsystems. However, two additional cells have been added to each battery to accommodate a two-cell failure (shorted).

The total load of 4.1 amperes, shared by the four batteries, results in an ampere-hour discharge for the longest eclipse period of 4-1/4 amperes x 1.15 hour, approximately equal to 1.2 ampere-hour per battery. Limiting the depth of discharge to 20 percent requires a 6.0 ampere-hour cell. At this depth of discharge, the end-of-discharge voltage will remain above 28.8 volts (1.2 volts per cell).

5.3 Battery Charging: Charging shall be adequate to fully charge the batteries in less than 24 hours. Battery charging is performed at the maximum rate of 300 milliamperes, as limited by the charge regulators. Maximum current available for battery charging with the electronics loads is 1.0 ampere.

The charge will be terminated upon reaching a fully-charged condition by battery regulator charge logic. Specially constructed cells containing sensory elements will have a slightly lower ampere-hour capacity than the other cells in the string. The device to sense a fully-charged condition is an auxiliary electrode in the cell. The sensory electrode is an oxygen electrode similar to those used in fuel cells. The output current of the electrode is proportional to the buildup of the partial pressure of oxygen in the cell as a fully-charged state is reached.

6.0 REGULATORS

6.1 Battery Charge Regulators: Battery charge regulators for SYNCOM II will be of a boost type. The 24-cell batteries used require charge battery terminal voltages in excess of 36 volts. The use of a boost type of charging regulator permits battery charging continuously from the solar panel, and at the same time minimizes the total number of series solar cells required, because solar panel design can be based on the minimum voltage input to the electronics subsystems (27.3 volts), rather than the high battery charge voltages required.

The use of this boost regulator results in a higher overall efficiency of the regulator, since only a fraction of the battery charging power must be transformed. The regulator senses the battery state-of-charge and regulates the current into the series battery string. Several battery sensory electrodes are connected to the regulator charge control circuitry with "or" gates to sense the highest cell charge and prevent battery overcharge.

Solar Cells Specification

1.0 SCOPE

1.1 This specification covers the requirements for the design and construction of a photovoltaic solar cell to be used on the Synchronous Communications Satellite (SYNCOM) MARK II solar panel assembly.

1.2 Design Objectives: The solar cell shall be designed to meet all electrical, optical, mechanical and environmental requirements as specified herein. Test programs shall be successfully completed demonstrating the ability of the solar cell to meet all performance requirements as required by this specification. The solar cell shall be designed for optimum operation in accordance with the following relative priority list:

a) Reliability

b) Air mass zero sunlight conversion efficiency

c) Spectral characteristics

d) Thermal characteristics

e) Weight

1.3 Conflicting Requirements: Conflicting requirements arising between this specification or of any specification or drawing listed herein shall be referred in writing to Hughes Aircraft Company (HAC) for interpretation and clarification.

1.3.1 Requests for Deviation: Requests for deviation from this specification, applicable drawings, specifications, publications, materials and processes specified herein, shall be considered design changes or design deviations and shall not be allowed except by written authorization from HAC.

1.4 Materials, Parts and Processes: When a material, part or process is not specified herein, the seller's selection shall assure the highest uniform quality and condition of the product, suitable for the intended use, and such selection shall be submitted for the review and concurrence of HAC.

2.0 APPLICABLE DOCUMENTS

2.1 The following documents of the date and/or revision shown are a part of this specification except as noted in subsequent paragraphs:

Military Specifications

MIL-STD-105 Sampling Procedures and Tables for Inspection by Attributes

Hughes Aircraft Company Specifications

225001 Quality Assurance, 17 May 1961

NASA Specifications

63-106 Specification for Determining Relative 1 Mev Electron Radiation Damage Resistance for Silicon Solar Cells, 31 October 1962

3.0 REQUIREMENTS

3.1 Design Description: The solar cell shall be a silicon "N on P" junction type, 1 by 2 centimeter size, the coverglass applied. The individual solar cells shall be covered with a coverglass capable of meeting the requirements specified herein. The solar cells will be capable of being bonded to a honeycomb substrate, and will be capable of being electrically interconnected in series-parallel groups to assure maximum reliability of operation.

3.1.1 Configuration: The dimensions and overall configuration of the solar cell shall be specified in the seller's drawing and shall be submitted for HAC approval.

3.1.2 Cell Defects: The maximum chip allowed shall be 0.010 inch deep by 0.150 inch long and the maximum corner crack shall be 0.045 inch on the hypotenuse.

3.1.2.1 Cell Covers: Each solar cell shall be covered with a 0.006 inch thick Corning Number 0211 microsheet glass coated with an anti-reflection coating on the top surface and an ultraviolet reflecting coating on the under surface. Cell to cover overlap and exposed active area shall not be greater than .005 inch.

3.1.2.2 Cell Cover and Adhesive Defects: Cracks, scratches, or discoloration will not be allowed. Chips will not extend more than 0.010 inch from an edge. There shall be no evidence of delamination, discoloration or bubbles in the adhesive. A maximum of five bubbles, none larger than 0.020 inch diameter, is acceptable per cover.

3.1.2.3 Solar Cell Absorptance and Emittance: Average emittance of the solar cell top surface with the coverglass applied shall not be less than 0.83 from 25°C to 125°C. Average absorptance to solar radiation in the wavelength region 0.2 to 2.5 microns shall not exceed 0.82.

3.1.2.4 Spectral Transmittance: The spectral transmittance of the cell cover shall meet the following requirements:

	Wavelength (Microns)	Transmittance Percent
	0.400 ± 0.015	50
From	0.300 to 0.370	Less than 1 average
From	0.500 to 1.000	92 minimum

3.1.3 Negative Contact: The exposed negative (top) contact of the cell shall have a clean, uniform and complete line of solder along the 2 cm dimension. The width of this solder line shall not be less than 0.032 inch.

3.1.4 Positive Contact: The positive contact of the cell shall be flat within 0.005 inch and free of all contaminating material. The surface shall be optimized so as to provide maximum adhesive bonding strength.

3.1.5 Contact Coverage: Solder coverage of "N" and "P" contacts shall be 90 percent minimum.

3.1.6 Weight: Total assembled solar cell weight including coverglass shall not exceed 0.35 gram average per lot.

3.2 Power Output: The power output of each solar cell with coverglass applied under air mass zero spectral conditions, and solar radiation intensity of 140 mw/cm^2 shall meet the following requirements:

Test Conditions		
Temperature °C	Voltage Volts	Power Milliwatts
25 ± 2	0.46 ± 0.010	22.7 minimum

The specified voltage shall be used in measuring the power output of the solar cells. The electrical performance of the solar cell shall be measured with an illuminated source as specified in Paragraph 3.2.1.

3.2.1 Illumination Source: The source of radiation used to illuminate the cell for purposes of confirming cell power Paragraph 3.2 shall be sunlight at the earth's surface at Table Mountain, California, or at other HAC-approved test sites with the following minimum sunlight conditions:

1) 100 mw/cm^2 illumination intensity

2) Five miles clear visibility

3) Minimum sky radiation - This shall be determined by the ratio of solar cell short circuit current under the conditions of uncollimated and collimated sunlight. The ratio shall be as follows:

$$\frac{I_{sc} \text{ (uncollimated sunlight)}}{I_{sc} \text{ (collimated sunlight)}} \leqslant 1.08$$

A collimating tube equipped with baffles shall be used and the tube shall have a minimum length to diameter ratio of 10.

The power output under the test conditions of Paragraph 3.2 and at 100 mw/cm^2 intensity shall be multiplied by the factor 1.21 to obtain the air mass zero power output. The test data obtained for each cell subjected to test shall be submitted to HAC concurrent with delivery of each lot.

3.2.2 Temperature Variations: The seller shall furnish the voltage-current characteristics curves of the solar cell for 0°, 75° and 125°C. The tests shall be run with a constant illumination source as specified in Paragraph 3.2.1 or equivalent.

3.2.3 Illumination Intensity Variations: The seller shall furnish the voltage-current characteristic curves of the solar cell at intensities of 100, 115, 130 and 150 milliwatts per square centimeter.

3.3 Storage: The solar cell as specified in Paragraph 3.1 with coverglass installed, shall be capable of meeting the requirements specified below:

a) The solar cell shall be capable of meeting all performance requirements after storage at a relative humidity of 50 percent maximum and at a temperature of 21°± 15°C for a period of 24 months.

b) The solar cell shall be capable of meeting all performance requirements after storage at a relative humidity of 95 percent maximum and at a temperature of 24° ± 20°C for a period of one month.

3.4 Radiation Damage Resistance: The seller shall provide evidence of compliance with the NASA-GSFC Radiation Damage Procurement Specification, Specification No. 63-106 dated 31 October 1962.

3.5 Environmental Performance: The solar cell shall meet all performance requirements of this specification after having been subject to the environmental conditions specified in Paragraph 4.0 of this specification.

3.6 Interchangeability: Solar cells bearing the same part number shall be physically and functionally interchangeable without selection or fit. The HAC part number for this solar cell shall be 170263.

4.0 TESTS

4.1 General

4.1.1 Test Apparatus: All meters, scales, thermometers, and similar measuring test equipment used in conducting tests specified herein shall be accurate within 1 percent of the full-scale value. Full-scale deflections of meters should not be more than twice the maximum value of the quantity being measured. All test apparatus shall be calibrated at suitable intervals and records of such calibration shall be available for inspection by Hughes. Hughes may examine the seller's test equipment and determine that the seller has available and utilizes correctly, gauging, measuring and test equipment of the required accuracy and precision, and that the instruments are of the proper type and range to make measurements of the required accuracy. The calibration of gauges, standards, and instruments will be checked in a mutually agreed upon primary standards laboratory if disputes concerning performance occur.

4.1.2 Test Records: Records shall be kept of all tests and of applicable manufacturing data and these records shall be made available for inspection by HAC. Prior to and following each test of Paragraph 4.6, a thorough visual examination of the test solar cell shall be conducted. All physical markings, defects, and other visual characteristics shall be noted and recorded as a portion of the test records.

4.1.3 Test Conditions: Unless otherwise specified herein, all tests shall be performed at the following nominal ambient conditions:

a) Temperature - 25°C

b) Barometric pressure - 29.92 inches of mercury

c) Relative humidity - not greater than 50 percent

4.2 Classification of Tests: Tests shall be classified as follows:

a) Acceptance Tests

b) Type Approval Tests

4.3 Sampling Procedure: The sampling procedure for acceptance tests of Paragraph 4.5 shall meet the requirements of Military Specification MIL-STD-105 for an AQL of 2.5 percent defective, excluding the Electrical Performance Tests of Paragraph 4.5.2.

4.4 Test Location: Unless otherwise specified in the contract, type approval and acceptance tests shall be performed by the seller at the seller's plant. If the use of outside test facilities are required, the use of these facilities shall be subject to approval by HAC. HAC shall have the right to witness, inspect, and review all type approval and acceptance tests.

4.5 Acceptance Tests: Samples of all lots of solar cells submitted for delivery shall be subjected to the acceptance tests listed below. A lot shall consist of 25,000 solar cells manufactured under essentially the same conditions and submitted for acceptance at substantially the same time. The sampling plan shall comply with Paragraph 4.3.

4.5.1 Examination of Product: The solar cell shall be inspected to determine compliance with respect to materials, workmanship, dimensions, and weight as specified in Paragraphs 3.1.1, 3.1.2, 3.1.3, 3.1.4, 3.1.5, and 3.1.6.

4.5.2 Electrical Performance: The power output of the solar cell shall be determined at a temperature of 25° ± 2°C. To comply with the requirements of Paragraph 3.2, 200 solar cells out of each 25,000 solar cell lot will be selected in a random manner and their electrical performance determined in sunlight as specified in Paragraph 3.2.1. The data obtained from this sunlight measurement will be employed to calibrate a laboratory light source and thereby establish acceptance criteria for the solar cells at the seller's and buyer's facility. In addition to seller's acceptance tests, Hughes will conduct electrical performance tests of delivered solar cells. Any cell determined to be defective during HAC inspection shall be cause for rejection of the entire lot. The light source used by the seller for the above testing shall have the approval of Hughes.

4.6 Type Approval Tests: Type approval tests shall be conducted in the manner described below and prior to final contract award. A sample of 100 solar cells with coverglass shall be selected at random from a production lot. When one or more test samples fails to meet the requirements of this specification, the extent and cause of failure shall be determined and corrective action initiated. After corrective action has been taken, type approval and acceptance tests shall be repeated as mutually agreed between Hughes and the seller upon review of the failure analysis. All cells subjected to type approval tests shall not be used for flight hardware. The solar cells shall be subjected to type approval tests in the order listed below.

4.6.1 Acceptance Tests: All solar cells shall be tested in accordance with and meet the requirements of Paragraph 4.5.

4.6.2 Electrical Performance Test: The power output of the solar cells shall be measured in accordance with Paragraph 3.2. Temperature of the solar cells shall be continuously monitored. The solar cells shall meet the requirements of Paragraph 3.2.

4.6.3 Storage Temperature and Humidity: The test specimens shall be placed in a sealed test chamber and the temperature and humidity raised during a two-hour period to 52°C and 95 percent relative humidity, respectively. At the end of a six-hour soak period, the heat source for the chamber will be turned off. During the following 16-hour period, the temperature shall drop at a uniform rate to 37°C or less. Three such 24-hour cycles shall be performed consecutively. At the end of this period, electrical performance tests in accordance with Paragraph 4.6.2 shall be conducted.

4.6.4 Temperature Cycling: The solar cells shall be subjected to five temperature cycles at a minimum thermal rate of 30°C per minute, between the extremes of 110° ± 2° and - 196 ± 2°C. The solar cells shall remain at the extremes for a minimum of one hour. Electrical performance tests in accordance with Paragraph 4.6.2 shall then be conducted.

4.6.5 High Temperature - Vacuum: The solar cells shall be placed in a test chamber and the chamber reduced in pressure to a vacuum of at least 10^{-5} Torr. The temperature shall be raised to 110° ± 2°C. The solar cells shall remain in the chamber for a period of 168 hours. At the end of this period, the solar cells shall be allowed to return to room ambient temperature and the electrical performance tests in accordance with Paragraph 4.6.2 shall be conducted.

4.6.6 Ultraviolet Radiation Tests: Fifteen of the 100 type approval solar cells shall be subjected to high intensity ultraviolet radiation from a Model No. 700-J Ultra-Violet Lamp Unit manufactured by Shannon Luminous Materials Company, Hollywood, California, or the equivalent. If the Shannon Lamp Unit is employed, no more than eight cells at a time shall be irradiated. The cells shall be positioned normal to the irradiation with the active cell areas facing the illuminating source. The cells shall be positioned about the centerline of the lamp unit at a distance of approximately 3-1/2 inches from the open end of the lamp housing. Forced air cooling shall be employed to maintain the cells at a temperature in the range 40° to 50°C. Duration of the test shall be 20 hours. Upon completion, the cells shall be tested for electrical performance in accordance with Paragraph 4.6.2.

4.6.7 Paragraph 3.1.2.2 shall apply after each test in Paragraphs 4.6.3, 4.6.4, 4.6.5 and 4.6.6.

4.7 Radiation Damage: In order to comply with Paragraph 3.4, the seller shall conduct the radiation damage tests in accordance with NASA-GSFC, Specification No. 63-106 or the seller shall provide sufficient evidence these tests have previously been completed satisfactorily. This test will not be considered a part of the type approval program but must also be completed prior to final contract award.

4.8 Retest: Any changes made in manufacturing techniques, processes, materials, quality control levels, manufacturing sites or type of manufacturing equipment shall be cause for complete retest per Paragraph 4.6 at no cost to HAC.

5.0 PREPARATION FOR DELIVERY

5.1 Shipping Container: The seller shall provide containers of the size required for the delivered lots with a desiccant capable of assuring container ambient relative humidities of no greater than 50 percent in compliance with the requirements of Paragraph 3.3.1(a). Desiccant may be replaced periodically if necessary. An indicator of desiccant water absorption should be provided.

5.2 Identification: Each solar cell shipping box shall be legibly identified by the following:

a) HAC part number

b) Seller's part number

c) Month and year of manufacture

d) Lot number

e) Solar cell serial number (1 through 25,000 for each lot)

6.0 QUALITY ASSURANCE PROVISIONS

6.1 General: The materials, processes and assembly covered by this specification shall be subject to extensive inspection and testing by both the seller and HAC.

6.2 Inspection

6.2.1 Seller Inspection: The seller shall establish a quality control system in accordance with or exceeding the requirements of HAC Specification 225001, Quality Assurance Specification. Product quality assurance shall be provided by the seller by a series of in-process inspections commencing with receipt of raw materials and parts and continuing through the finished product. The selected inspection points shall have the approval of Hughes. A record shall be maintained of all inspections and be subject to review by Hughes.

6.2.2 HAC Source Inspection: The Hughes Aircraft Company shall at its option provide inspection to adequately monitor the seller's quality control effort including in-process inspection and in-process tests. The completed hardware may be source inspected by HAC to assure that the product conforms to all the requirements specified on the applicable drawings and specifications and may include witnessing of acceptance tests.

6.2.3 Rejected Assemblies: Rejected assemblies shall not be resubmitted for approval without furnishing full particulars concerning the rejection, the measures taken to overcome the defects, and the prevention of their future occurrence. Each rejected assembly shall be identified by a serialized rejection tag. This rejection tag shall not be removed until rework requirements have been complied with, and then the tag shall be removed only by, or in the presence of, an authorized representative of HAC.

Battery Cell Specification (Standard Tube Terminal)

1.0 SCOPE

1.1 This specification covers a hermetically sealed nickel-cadmium battery cell to be used in the assembly of batteries for space applications.

2.0 APPLICABLE DOCUMENTS

None

3.0 REQUIREMENTS

3.1 Design Description: The cell shall be hermetically sealed nickel-cadmium type suitable for space application as specified herein.

3.1.1 Weight: The weight shall not exceed 0.65 pound.

3.1.2 Terminals: All electrode terminals shall be insulated from the case and contain provisions for solder-type connections of the lead wires.

3.1.3 Container: The cell container shall be capable of maintaining its original dimensions for the life of the battery under the storage and operating conditions specified herein.

3.1.4 Corrosion Resistance: All external surfaces of the cell shall show no evidence of corrosion when exposed to the environmental conditions specified herein.

3.1.5 Leakage: The cell shall show no signs of electrolyte leakage when subjected to the storage and operating conditions specified herein. The cell shall show no signs of leakage when tested in accordance with Paragraph 4.3.2.

3.1.6 Interchangeability: All cells having the same part number shall be functionally and dimensionally interchangeable.

3.1.7 Cell Marking: The following information shall be marked by stamping, etching or other suitable methods which will insure permanent legibility:

a) HAC part number

b) Serial number

c) Manufacturer's name, trademark, or code symbol

d) Terminal identification

3.2 Performance Requirements

3.2.1 Capacity: The cell discharge capacity at 75°F shall be a minimum of 6.0 ampere-hours when discharged at a constant current of 1.2 amperes to an end voltage of 1.0 volt. The voltage for 3.5 hours of the discharge period shall be 1.16 volts minimum.

The charge schedule to determine the ampere-hours discharge capacity and voltage requirements of this paragraph and paragraphs 3.2.2 through 3.2.6 shall be a constant current charge at 0.5 amperes for 16.0 hours followed by an open-circuit period of 1.0 hour. The charge shall be from a point of previous discharge to 1.0 volt at 1.2 ampere-hours.

3.2.2 Capacity at Low Temperature: With the cell case temperature maintained at 30°F during charging and discharging, the discharge capacity shall not be less than 4.8 ampere-hours when charged and discharged for the periods and rates specified in Paragraph 3.2.1.

3.2.3 Capacity at High Temperature: With the cell case temperature maintained at 100°F during charging and discharging, the discharge capacity shall not be less than 4.8 ampere-hours when charged and discharged for the periods and rates specified in Paragraph 3.2.1.

3.2.4 Capacity at High Rate Discharge: The cell discharge capacity at 75°F shall be a minimum of 4.8 ampere-hours when discharged at a constant current of 6.0 amperes to an end voltage of 1.0 volt.

The charge schedule used to meet the requirements of this paragraph shall be the same as the charge schedule specified in Paragraph 3.2.1.

3.2.5 High Current Discharge: The cell terminal voltage during discharge of a fully-charged cell at a load of 12.0 amperes shall be 1.0 volt minimum for a period of 10 seconds.

3.2.6 Overcharging Rate: With the cell case temperature maintained at 75°F, the cell shall be capable of withstanding a continuous overcharging current 0.5 ampere for a period of 30 days.

3.2.7 Maximum Charge Voltage: The maximum on-charge cell voltage shall not exceed 1.48 volts when meeting the requirements of Paragraph 3.2.6.

3.2.8 Charge Retention: The cell discharge capacity in ampere-hours, shall not be less than 80 percent of its initial capacity when discharged 30 days after being fully charged.

The cell shall meet the provisions of this paragraph when charged and discharged at the rates and periods specified in Paragraph 3.2.1. During the 30 days stand time, the cell case temperature will be maintained at 75° F.

3.2.9 Charge Retention at Minimum Charge: The cell open-circuit voltage shall be 1.16 volts minimum after 24 hours stand time when tested in accordance with Paragraph 4.3.4.

3.2.10 Charging at Minimum Rate: When charged at a constant current of 0.060 ampere for a period of 200 hours at 75° F, the cell discharge capacity shall not be less than 5.4 ampere-hours. The capacity shall be measured by discharging at a constant current of 1.2 amperes to an end voltage of 1.0 volt. Prior to discharge, the cell shall stand on open-circuit for a period of one hour.

3.2.11 Capacity After Cycling: The cell discharge capacity shall be a minimum of 4.8 ampere-hours after 500 charge-discharge cycles (at 75° F) of 20 percent depth with a 1.2 ampere discharge rate and 0.3 ampere charge rate. In addition, the discharge voltage shall be a minimum of 1.20 volts during the 500 discharge cycles.

3.3 Environmental Requirements

3.3.1 Storage: Each cell shall be capable of meeting all the requirements of this specification after storage for two years at any temperature between 20°F and 130°F.

3.3.2 Vacuum: Each cell shall be capable of meeting all the requirements of this specification in a vacuum environment of at least 10^{-10} mm Hg.

3.3.3 Humidity: Each cell shall be capable of meeting all the requirements of this specification after being subjected to a test chamber temperature of 130°F and a relative humidity of 95 percent for eight hours.

3.3.4 Thermal Shock: Each cell shall be capable of meeting all the requirements of this specification after being subjected to a test chamber temperature of -20° F for at least six hours immediately followed by exposure to a test chamber temperature of 150° F for at least six additional hours.

3.3.5 Shock: Each cell shall be capable of meeting all the requirements of this specification after being subjected to two 60g terminal peak sawtooth shock pulses of 15 millisecond duration each in each direction along the three principal cell axes.

3.3.6 Acceleration: Each cell shall be capable of meeting all the requirements of this specification after being subjected to steady accelerations of 30g for 60 seconds duration in each direction along the three principal cell axes.

3.3.7 Spin: Each cell shall be capable of meeting all the requirements of this specification while being spun continuously in any attitude at 140 rpm from a 26-inch radius.

3.3.8 Vibration: Each cell shall be capable of meeting all the requirements of this specification after being subjected to the vibration environment listed below along the three principal cell axes.

a. Sinusoidal Excitation

Frequency cps*	Duration, minutes	Level
5 - 15	4.3	0.25 in. double amplitude
15 - 250	4.3	3.0g (0 - peak)
250 - 400	4.3	5.0g (0 - peak)
400 - 2000	4.3	7.5g (0 - peak)

*log sweep at two octaves/minute.

b. Random Excitation

Frequency, cps	Duration, minutes	Level
20 - 80	6.0	$0.04g^2$/cps
80 - 1280	6.0	Increasing from $0.04g^2$/cps at 1.22 db/octave
1280 - 2000	6.0	$0.07g^2$/cps

4.0 TESTS

4.1 General

4.1.1 Test Apparatus: All meters, scales, thermometers, and similar measuring test equipment used in conducting tests specified herein shall be accurate within one percent of the full-scale value. Full-scale deflections of meters should not be more than twice the maximum value of the quantity being measured. Periods of discharge and charge shall be timed with a device accurate within 0.2 percent. All test apparatus shall be calibrated at suitable intervals against standards traceable to the National Bureau of Standards. Records of such calibration shall be available for inspection.

4.1.2 Records: Records shall be kept and be made available for inspection of the tests and of applicable manufacturing data (e.g., serial numbers of batteries manufactured from each lot of raw or processed material).

4.1.3 Test Conditions: Unless otherwise stated, laboratory ambient conditions of tests shall be:

a) Temperature 70 ± 10°F

b) Barometric pressure 30 ± 2 inches of Mercury

c) Relative humidity, less than 90 percent

4.1.4 Tolerances: Unless specifically stated in the test procedures, the following test tolerances are allowable:

a) Ambient temperature	± 5°F
b) Relative humidity	± 5 percent
c) Vibration level	± 10 percent
d) Pressure	± 5 percent
e) Frequency	± 2 percent
f) Shock	± 10 percent
g) Acceleration	± 10 percent

4.1.5 Rejections and Retest: When one or more cells from a lot fails to meet the requirements of this specification in a manner indicative of a systematic design deficiency, acceptance of all items in the lot will be withheld until the extent and cause of the failure is determined and corrective action initiated. A lot shall consist of cells manufactured essentially under the same conditions, from the same materials stock, and at the same time. After corrective action has been taken, acceptance and qualification tests shall be repeated as mutually agreed between Hughes Aircraft Company and the cell manufacturer upon review of the failure analysis. Cells, which have been rejected, may be reworked or replaced to correct any defects and re-submitted for acceptance. Before re-submitting the cells for test, full particulars concerning the rejection and corrective action taken shall be furnished to Hughes Aircraft Company. If investigation of a test failure indicates that defects may exist in cells already accepted, these cells shall be retested and reworked or replaced as required to comply with this specification. Cells which fail to meet specific selection or acceptance test requirements shall be rejected on an individual cell basis.

4.1.6 Additional Tests: Additional tests shall be conducted by HAC as deemed necessary to verify that the cell can meet the requirements of this specification. These tests shall not impose more stringent requirements than those specified in this specification. Failure of the cell to pass these additional tests shall be cause for rejection in accordance with Paragraph 4.1.5.

4.2 Classification of Tests: Tests shall be classified as follows:

a) Acceptance Tests

b) Qualification Tests

4.3 Acceptance Tests: All cells submitted for delivery shall be subjected to the following tests. These tests shall be conducted at laboratory ambient conditions. Upon completion of each test, specimens and test data shall be examined to determine compliance with this specification.

4.3.1 Examination of Product: Each cell shall be inspected to determine compliance with respect to material, workmanship, dimensions, weight, and product marking.

4.3.2 Leakage Test: A leakage test shall be conducted on each cell by one of the two methods described below:

a) Helium Leak Tests (for cells containing Helium gas): The cell shall be placed in a vacuum chamber and the pressure reduced to at least 10^{-4} mm Hg and maintained for at least five minutes. The helium leakage rate from the cell shall be measured with a Consolidated Electrodynamic Corporation, Model 24-120 leak detector or equivalent. The leakage rate shall not exceed one cubic centimeter of helium per month.

b) Electrolyte Indicator Leak Test: In lieu of the helium leak detection method, an electrolyte indicator test may be used. The indicator shall be a one percent solution of phenolphthalein in alcohol or P-H indicator paper. Any change in color of the indicator shall be evidence of electrolyte leakage.

4.3.3 Capacity Discharge Test: Each cell shall be charged per Paragraph 3.2.1, allowed to stand on open circuit for one hour, and then discharged at a constant current of 1.20 amperes to an end voltage of 1.0 volt. This test shall be repeated for a second charge-discharge cycle. Each cell shall meet the discharge capacity and voltage requirements of Paragraph 3.2.1.

4.3.4 Charge Retention - Minimum Charge: Following the discharge of Paragraph 4.3.3, the cell shall be short circuited for 12 hours minimum. The short circuit shall be removed and the cell charged at a constant current of 0.5 ampere for 10 minutes. The cell shall then be placed on open circuit for 24 hours during which time the open circuit voltage of the cell shall be 1.16 volts minimum. (Reference, Paragraph 3.2.9.)

4.3.5 Overcharge: Following the test of Paragraph 4.3.4, the cell shall be charged at 0.5 ampere for a period of 96 hours. The cell on-charge voltage shall not exceed 1.48 volts. Following this test, the cell shall again be subjected to the leakage test and meet the requirements of Paragraph 4.3.2.

4.4 Qualification Tests: Battery cells submitted for qualification tests shall be typical of production line batteries of the final design for flight usage. A minimum of 20 cells shall be subjected to the tests of Paragraph 4.4.1 through 4.4.12 and a minimum of 20 cells for the test of Paragraph 4.4.1 through 4.4.5 and 4.4.13. The tests on each cell shall be conducted in the order listed below. All cells subjected to qualification testing shall meet all requirements herein.

4.4.1 Acceptance Tests: All cells submitted for qualification tests shall be tested in accordance with and meet the requirements of Paragraph 4.3.

4.4.2 Thermal Shock Test: The cells, after being fully charged, shall be subjected to a test chamber temperature of -20° F for a period of six hours followed immediately by exposure to a test chamber temperature of +150° F for an additional six hours. Open circuit voltage of each cell shall be 1.25 volts minimum following this test.

4.4.3 Vibration Test: Following the test of Paragraph 4.4.2, the cells shall be mounted rigidly to a test fixture and subjected to the vibration environment in each of the three orthogonal axes as shown below. Open circuit voltage of each cell shall be 1.25 volts minimum following this test.

a. Sinusoidal Excitation

Frequency cps*	Duration, minutes	Level
5 - 15	4.3	0.25 in. double amplitude
15 - 250	4.3	3.0g (0 - peak)
250 - 400	4.3	5.0g (0 - peak)
400 - 2000	4.3	7.5g (0 - peak)

*Log sweep at two octaves/minute

b.

Frequency, cps	Duration, minutes	Level
20 - 80	6.0	$0.04g^2/cps$
80 - 1280	6.0	Increasing from $0.04g^2/cps$ at 1.22 db per octave
1280 - 2000	6.0	$0.77g^2/cps$

4.4.4 Shock Test: Following the test of Paragraph 4.4.3, the cells shall be mounted rigidly in test fixture and subjected to three 60 g terminal peak sawtooth shock pulses of 15 milliseconds duration each in each direction along the three principal cell axes. The open circuit voltage of each cell shall be 1.25 volts minimum following this test.

4.4.5 Acceleration Test: Following the test of Paragraph 4.4.4, the cells shall be subjected to steady accelerations of 30 g for 60 seconds duration in each direction along the three principal cell axes. The open circuit voltage of each cell shall be 1.25 volts minimum following this test.

4.4.6 Spin Test: Following the test of Paragraph 4.4.5, each cell shall be discharged to 1.0 volts. The cells shall then be spun in a test fixture at 208 rpm from a radius of 26 ± 1 inch. The cells shall be oriented such that the longitudinal axis of the cell is along the radius vector of rotation and the cell terminals face the center of rotation. While spinning the cells shall be charged at 0.5 amperes for 16.0 hours, placed on open circuit for 1.0 hour and then discharged at a constant current of 1.2 amperes to an end voltage of 1.0 volts. Each cell shall meet the discharge capacity and voltage requirements of Paragraph 3.2.1.

4.4.7 Low Temperature Capacity Test: Following the test of 4.4.6, the cells shall be placed in a temperature chamber and maintained at 30° F throughout this test. The cells shall be charged at 0.5 ampere for 16.0 hours, placed on open circuit for 1.0 hour and then discharged at a constant current of 1.2 amperes to an end voltage of 1.0 volt. The discharge capacity shall not be less than 4.8 ampere-hours for each cell. (Ref. Paragraph 3.2.2.)

4.4.8 High Temperature Capacity Test: Following the test of Paragraph 4.4.7, the cells shall be placed in a temperature chamber and maintained at 100°F throughout this test. The cells shall be charged at 0.5 amperes for 16.0 hours, placed on open circuit for 1.0 hour and then discharged at a constant current of 1.2 amperes to an end voltage of 1.0 volts. The discharge capacity shall not be less than 4.8 ampere-hours for each cell. (Ref. Paragraph 3.2.3.)

4.4.9 Capacity at High Rate Discharge: Following the test of Paragraph 4.4.8, the cells shall be placed in a laboratory temperature environment of 75° F. The cells shall be charged at 0.5 ampere for 16.0 hours, placed on open circuit for 1.0 hour and then discharged at a constant current of 6.0 amperes to an end voltage of 1.0 volts. The discharge capacity shall not be less than 4.8 ampere-hours for each cell. (Ref. Paragraph 3.2.4.)

4.4.10 Charging at Minimum Rate Test: Following the test of Paragraph 4.4.9, the cells shall be charged at a constant current of 0.060 ampere for a period of 200 hours, placed on open circuit for one hour and then discharged at a constant current of 1.2 amperes to an end voltage of 1.0 volts. The discharge capacity shall not be less than 5.4 ampere-hours. (Ref. Paragraph 3.2.10.)

4.4.11 High Current Discharge Capability Test: Following the test of Paragraph 4.4.10, the cells shall be fully charged, placed on open circuit for one hour, then discharged at a rate of 12.0 amperes for a period of 10 seconds. The voltage for the 10-second discharge period shall be 1.0 volts minimum. (Ref. Paragraph 3.2.5.)

4.4.12 Overcharge Test: Following the test of Paragraph 4.4.11, the cells shall be continuously overcharged at a constant current of 0.5 ampere for a period of 30 days. During this period the cells shall be in a 75° F temperature environment. The cell on-charge voltage shall not exceed 1.48 volts during the 30-day period. (Ref. Paragraph 3.2.6.)

4.4.13 Charge Retention Test: Following the test of Paragraph 4.4.12, the basic capacity of the cells shall be redetermined in accordance with the test procedure of Paragraph 4.3.3. The cells shall then be charged to full capacity and placed on open circuit for a period of 30 days. At the end of 30 days, the discharge capacity shall not be less than 80 percent of the initial discharge capacity measured just prior to the 30-day period (Ref. Paragraph 3.2.8).

4.4.14 Cycle Test: After the cells have completed the tests of Paragraph 4.4.1 through 4.4.6, they shall be subjected to the following cycle testing:

Cycles	Charge Current	Charge Time	Discharge Current	Discharge Time	Ambient Temperature
1 - 500	0.3 amp	7.0	1.2 amp	1.0 hour	75°F

At the end of each 100 cycles the basic ampere-hour capacity to 1.0 volts shall be determined in accordance with the test procedure of Paragraph 4.3.3. At the end of 500 cycles, the discharge capacity shall not be less than 4.8 ampere-hours. In addition, the end-of-discharge voltage during the 500 cycles shall not be less than 1.2 volts. (Ref. Paragraph 3.2.11.)

Battery Cell Specification (Full-Charge Sensory Electrode)

1.0 SCOPE

1.1 This specification covers a hermetically sealed nickel-cadmium battery cell which contains a sensory oxygen electrode to indicate when the cell reaches a fully-charged state. This cell will be used in series with cells described in Hughes Procurement Specification No. X30630-001 in the assembly of batteries for space vehicle usage.

2.0 APPLICABLE DOCUMENTS

Hughes Procurement Specification No. X30630-001, Sealed Nickel-Cadmium Battery Cell, 6.0 ampere-hours.

3.0 REQUIREMENTS

In addition to the requirements of this specification, the cells shall meet all the requirements of Hughes Specification X30630-001 without utilizing the oxygen electrode.

3.1 Performance Requirements of the Sensory Electrode

3.1.1 Electrical Output: With the cell being charged at any rate between 0.060 ampere and 0.5 ampere and a 1.5 ohm external resistance between the oxygen electrode and the negative electrode of the battery; the indication that the cell has reached a fully-charged state shall be an increase of potential difference between the oxygen electrode and negative electrode to 0.9 volt minimum. During charging, when the cell is in a state less than fully charged, the potential difference between the oxygen electrode and the negative electrode shall be less than 0.3 volt.

3.1.2 Minimum Cell Performance at Maximum Full Charge Indication: The requirements of HAC Specification X30630-001, Paragraphs 3.2.1, 3.2.2, 3.2.3, and 3.2.10 shall also be met by charging the cell at the prescribed rate for each test until the potential difference between the oxygen electrode and the negative electrode reaches 0.9 volt. At this point the charge shall be terminated and the cell tested in accordance with the remaining provisions of the paragraph. This requirement shall be met with a maximum external resistance of 1.5 ohms between the oxygen electrode terminal and the negative electrode terminal.

3.1.3 Minimum Cell Performance at an Intermediate Charge Indication: The cell shall meet the ampere-hour requirements of HAC Specification X30630-001, Paragraphs 3.2.1, 3.2.2, 3.2.3, and 3.2.10

diminished by 10 percent, by charging the cell at the prescribed rate for each test until the potential difference between the oxygen electrode and the negative electrode reaches 0.3 volt. At this point the charge shall be terminated and the cell tested in accordance with the remaining provisions of the paragraph. This requirement shall be met with a minimum external resistance of 25 ohms between the oxygen electrode terminal and the negative electrode terminal.

3.1.4 Charge Indication During Discharge: During discharge from a fully-charged state, the 0.9 volt indication between the oxygen electrode and the negative electrode shall decrease to below 0.3 volt before 15 percent of the ampere-hour capacity of the cell is discharged. This requirement shall be met with a minimum external resistance of 25 ohms between the oxygen electrode terminal and the negative electrode terminal. This requirement shall be met coincidentally with ampere-hour discharge requirements of HAC Specification X30630-001, Paragraphs 3.2.1, 3.2.2, 3.2.3, and 3.2.10.

4.0 TESTS

In addition to the tests of this specification, the cells shall be subjected to and meet the test requirements of Sections 4.1, 4.2, 4.3, and 4.4 of HAC Procurement Specification X30630-001 as specified herein.

4.1 Acceptance Tests: In addition to the Acceptance Tests of Section 4.3 of HAC Procurement Specification X30630-001, the following tests shall be performed on all cells submitted for delivery.

4.1.1 Cell Capacity Test at Full Charge Indication (75°F): With a 1.5 ohm external resistance between the oxygen electrode and the negative electrode of the battery, the cell shall be charged at 0.5 ampere constant current until the potential difference between the oxygen electrode and the negative electrode reaches 0.9 volt. At this point, the charge shall be terminated and the cell allowed to stand on open circuit for one hour. The cell shall then be discharged at a constant current of 1.20 amperes to an end voltage of 1.0 volts. This test shall be repeated for a second charge-discharge cycle. Each cell shall demonstrate a discharge capacity for each cycle of 6.0 ampere-hours. In addition, the voltage for 3.5 hours of the discharge period shall be 1.16 volts minimum. This test shall be performed in a 75°F environment.

4.2 Qualification Tests: Four cells typical of production line batteries of the final design for flight usage shall be subjected to all the qualification tests of Section 4.4 of Hughes Procurement Specification X30630-001 excepting for Paragraph 4.4.14. In addition, the same cells shall be subjected to the qualification tests of Paragraphs 4.2.1 through 4.2.8 of this specification. Four additional cells shall be subjected to the qualification tests of Paragraphs 4.4.1 through 4.4.6 of Hughes Procurement Specification X30630-001. In addition, the same cells shall be subjected to the qualification test of Paragraph 4.2.9 of this specification.

4.2.1 Cell Capacity Test at Full Charge Indication (30° F): This test shall be performed in a test chamber maintained at 30° F. With a 1.5 ohm external resistance between the oxygen electrode and the negative electrode of the battery, the cell shall be charged at 0.5 ampere constant current until the potential difference between the oxygen electrode and the negative electrode reaches 0.9 volt. At this point, the charge shall be terminated and the cell allowed to stand on open circuit for one hour. The cell shall then be discharged at a constant current of 1.20 amperes to an end voltage of 1.0 volt. The discharge capacity shall be 4.8 ampere-hours minimum. In addition, the 0.9 volt indication shall decrease to below 0.3 volt before 15 percent of the ampere-hour capacity of the cell is discharged.

4.2.2 Cell Capacity Test at Full Charge Indication (100° F): This test shall be performed in a test chamber maintained at 100° F. With a 1.5 ohm external resistance between the oxygen electrode and the negative electrode of the battery, the cell shall be charged at 0.5 ampere constant current until the potential difference between the oxygen electrode and the negative electrode reaches 0.9 volt. At this point the charge shall be terminated and the cell allowed to stand on open circuit for one hour. The cell shall then be discharged at a constant current of 1.20 amperes to an end voltage of 1.0 volt. The discharge capacity shall be 4.8 ampere-hours minimum. In addition, the 0.9 volt indication shall decrease to below 0.3 volt before 15 percent of the ampere-hour capacity of the cell is discharged.

4.2.3 Cell Capacity Test at Full Charge Indication at Minimum Charge Rate: This test shall be performed at 75°F ambient temperature. With a 1.5 ohm external resistance between the oxygen electrode and the negative electrode of the battery, the cell shall be charged at 0.060 ampere constant current until the potential difference between the oxygen electrode and the negative electrode of the battery reaches 0.9 volt. At this point the charge shall be terminated and the cell allowed to stand on open circuit for one hour. The cell shall then be discharged at a constant current of 1.2 amperes to an end voltage of 1.0 volts. The discharge capacity shall be 5.4 ampere-hours minimum. In addition, the 0.9 volt indication shall decrease to 0.3 volt before 15 percent of the ampere-hour capacity is discharged.

4.2.4 Cell Capacity Test at Intermediate Charge Indication (75°F): With a 1.5 ohm external resistance between the oxygen electrode and the negative electrode of the battery, the cell shall be charged at 0.5 ampere constant current until the potential difference between the oxygen electrode and the negative electrode reaches 0.9 volt. At this point, the charge shall be terminated and the cell allowed to stand on open circuit for one hour. The cell shall then be discharged at a constant current of 1.20 amperes to an end voltage of 1.0 volt. This test shall be repeated for a second charge-discharge cycle. Each cell shall demonstrate a discharge capacity for each cycle of 6.0 ampere-hours. In addition, the voltage for 3.5 hours of the discharge period shall be 1.16 volts minimum. This test shall be performed in a 75°F environment.

4.2.5 Cell Capacity Test at Intermediate Charge Indication (30° F): This test shall be performed in a test chamber maintained at 30° F. With a 1.5 ohm external resistance between the oxygen electrode and the negative electrode of the battery, the cell shall be charged at 0.5 ampere constant current until the potential difference between the oxygen electrode and the negative electrode reaches 0.9 volts. At this point, the charge shall be terminated and the cell allowed to stand on open circuit for one hour. The cell shall then be discharged at a constant current of 1.20 amperes to an end voltage of 1.0 volt. The discharge capacity shall be 4.8 ampere-hours minimum. In addition, the 0.9 volt indication shall decrease to below 0.3 volt before 15 percent of the ampere-hour capacity of the cell is discharged.

4.2.6 Cell Capacity Test at Intermediate Charge Indication (100° F): This test shall be performed in a test chamber maintained at 100°F. With a 1.5 ohm external resistance between the oxygen electrode and the negative electrode of the battery, the cell shall be charged at 0.5 ampere constant current until the potential difference between the oxygen electrode and the negative electrode reaches 0.9 volt. At this point, the charge shall be terminated and the cell allowed to stand on open circuit for one hour. The cell shall then be discharged at a constant current of 1.20 amperes to an end voltage of 1.0 volt. The discharge capacity shall be 4.8 ampere-hours minimum. In addition, the 0.9 volt indication shall decrease to below 0.3 volt before 15 percent of the ampere-hour capacity of the cell is discharged.

4.2.7 Cell Capacity Test at Intermediate Charge Indication and Minimum Charge Rate: This test shall be performed at 75° F ambient temperature. With a 1.5 ohm external resistance between the oxygen electrode and the negative electrode of the battery, the cell shall be charged at 0.060 amperes constant current until the potential difference between the oxygen electrode and the negative electrode of the battery reaches 0.9 volt. At this point the charge shall be terminated and the cell allowed to stand on open circuit for one hour. The cell shall then be discharged at a constant current of 1.2 amperes to an end voltage of 1.0 volt. The discharge capacity shall be 5.4 ampere-hours minimum. In addition, the 0.9 volt indication shall decrease to 0.3 volt before 15 percent of the ampere-hour capacity is discharged.

4.2.8 Charge Indication During Discharge (Maximum Impedance): With a 25-ohm external resistance between the oxygen electrode and the negative electrode of the battery, the cell shall be charged at 0.5 ampere constant current until the potential difference between the oxygen electrode and the negative electrode reaches 0.9 volt. At this point the charge shall be terminated and the cell shall be discharged immediately at 1.20 amperes to an end voltage of 1.0 volt. During discharge the 0.9 volt indication between the oxygen electrode and the negative electrode shall decrease to below 0.3 volt before 15 percent of the ampere-hour capacity of the cell is discharged.

4.2.9 Cycle Test: After the cells have completed the tests of Paragraph 4.4.1 through 4.4.6 of the Hughes Procurement Specification X30630-001, they shall be subjected to the following cycle testing:

Cycles	Charge Current	Charge Time	Discharge Current	Discharge Time	Ambient Temperature
1 - 500	0.3 amp	to 0.9 volt indication on sensory electrode	1.2 amps	1.0 hour	75° F

At the end of each 100 cycles the ampere-hour capacity shall be determined in accordance with the test procedure of Paragraph 4.1.1. At the end of 500 cycles, the discharge capacity shall not be less than 4.8 ampere-hours. In addition, the end-of-discharge voltage during the 500 cycles shall not be less than 1.2 volts.

Battery Regulators

The Advanced SYNCOM battery regulators possess the following capabilities:

1. Compatibility with the solar array impedance characteristics and those of the unregulated bus.

2. Provide necessary switching of the battery to the unregulated bus to permit satisfactory spacecraft operation under eclipse and transient load periods.

3. Provide current limiting to the battery charging network to eliminate any excessive unregulated bus drain in the event of a battery failure.

4. Enable the solar array to charge the battery regardless of maximum normal electronics operation. This is opposed to SYNCOM I, which requires turning off the electronics in order to charge the batteries.

5. To provide satisfactory operation with or without battery cells containing a sensory electrode.

Two different types of battery charging regulators have been breadboarded and are under test. The test setups are shown in Figure 6-29a and 6-29b.

Both circuits are of the "boost-add" type in that a controlled incremental voltage (10V) is added to the unregulated bus voltage (28V) to provide sufficient potential to charge the batteries.

One circuit utilizes a silicon controlled rectifier (SCR) in a standard Morgan chopper circuit (see Figure 6-29c. The SCR is turned on by use of a unijunction transistor at approximately a 2 K cps rate. The SCR is turned off by the discharge of a capacitor through the toroidal switching transformer after saturation.

The second circuit under consideration is a standard two-transistor static inverter with transformer feedback for positive control (see Figure 6-29d) of switching.

Both circuits will be refined for the next 60 days at which time the final circuit type will be chosen.

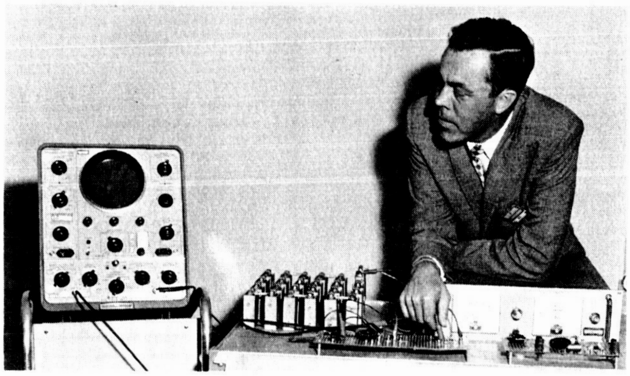
Apogee Injection Rocket Motor

The Syncom II apogee engine is similar to the JPL-developed Syncom I apogee engine with respect to configuration, materials, and propellant formulation. The engine will provide a velocity increment of 6100 feet per second for an injected spacecraft weight of 1518 pounds. An offloading capability commensurate with a spacecraft weight of 1300 pounds has been incorporated into the design.

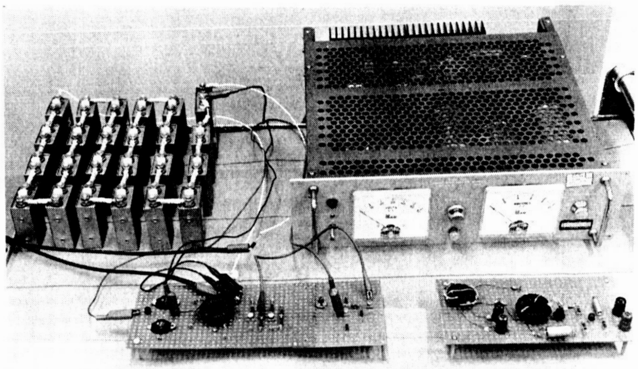
Development Program Progress

Heavy-weight engine cases, currently on order, are due during the week of 29 April 1963 at JPL; flight-weight cases are to be delivered during August 1963 for developmental tests. Heavy-weight truncated conical nozzles are on order, and contoured flight nozzles are in the late design phase. The initial heavy-weight test will be conducted during June 1963, and the initial flight-weight test during September of the same year.

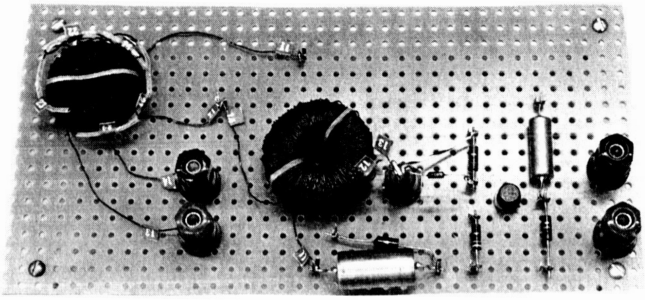
To date, three of the four planned subscale tests have been conducted, utilizing Syncom I engine components. The purpose of these tests was to evaluate performance at simulated altitude conditions with conical nozzles and to ensure adequate performance of the new altitude cell at Edwards AFB. The test program has been successful and the altitude simulation (HYPROX) system operated satisfactorily.



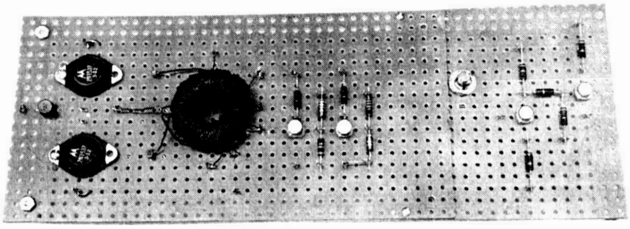
a) Under test



b) Closeup



c) SCR charge regulator



d) Transistor charge regulator cell

Figure 6-29. Battery Charge System Breadboard

Installation of the 150-gallon mixer at Edwards Air Force Base, which is to be used for loading Syncom II engines, is proceeding on schedule. Mixer operation will be turned over to JPL during September 1963.

STRUCTURE

Structural Design

The ATD spacecraft, T-1, was completed during this report period and delivered to the environmental test facility, where vibration tests were begun.

The following figures show views of the spacecraft structure during final assembly. Figure 6-30a is a top view of the aft subassembly. The circle of apogee motor brackets can be seen inside the thrust tube. A closeup of these brackets is shown in Figure 6-30b. Figures 6-30c and 6-30d show the forward and center subassemblies joined together, ready for attachment to the aft subassembly. In these views can be seen the bipropellant tanks, one velocity rocket, two altitude and spin control rockets, and the four sun sensor clusters. Strain gauges, installed for the vibration test, can be seen in numerous locations on the forward trusses.

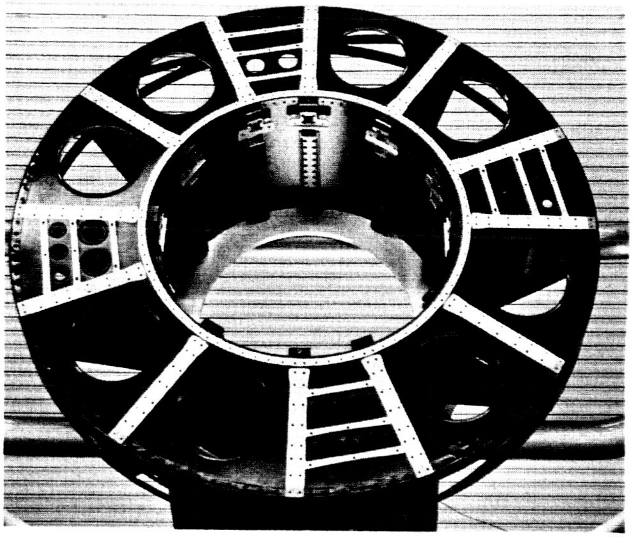
Figure 6-30e is a closeup of a mockup fuel tank and velocity control rocket. In an attempt to simulate the effect of fuel sloshing during the dynamic testing the tanks are loaded with the correct weights of two liquids (xylene and trichlorethylene) that closely resemble the fuel and oxidizer in density and viscosity.

Figure 6-30f shows the inert apogee motor mounted on its handling and support ring. The motor was built by Hughes to closely resemble JPL's motor design and was loaded by JPL with inert propellant.

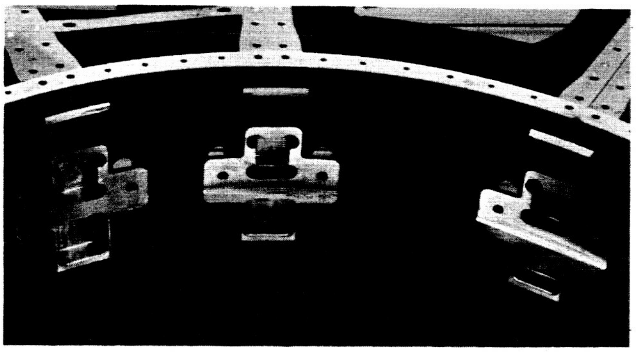
Marquardt Center Section. During the first portion of April, the additional center subassembly was completed and delivered to the Marquardt Corporation for installation of their developmental model bipropellant control unit. Figure 6-30g shows a top view of this assembly, which includes a truss to mount one altitude and spin-control rocket. In the aft view, Figure 6-30h, is seen the 3/8-inch plate bulkhead which in the absence of the aft subassembly is required to give structural rigidity and integrity to this partial assembly.

Modifications under Consideration. The outline dimensions of the spacecraft remain essentially unchanged from the previous report and the major structural subdivisions are retained. To provide increased heat conductance through the structure, the magnesium alloy parts are being replaced with aluminum alloy parts. An effort is being made to provide improved fabrication and maintenance access to the spacecraft components.

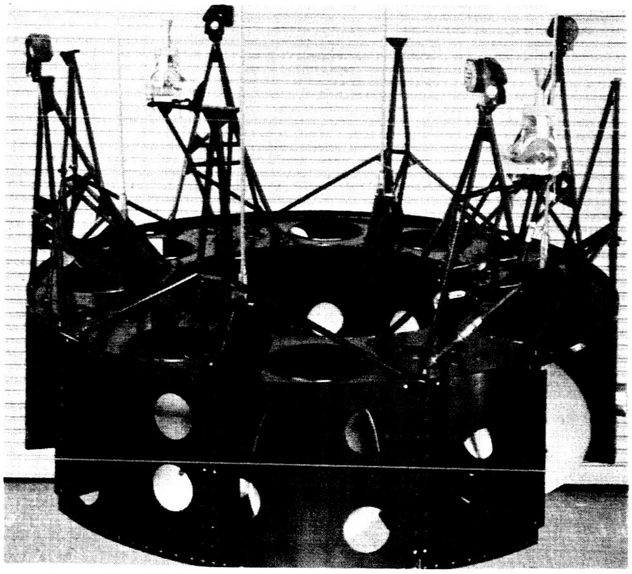
The aft thrust tube may be reduced from a 30-inch diameter to a 28.7-diameter, to eliminate the load path eccentricity between the motor attach diameter and the thrust support structure. All longitudinal stiffening members would be placed on the outer surface of the thrust tube to provide shell stiffness and provide attachment for component support brackets.



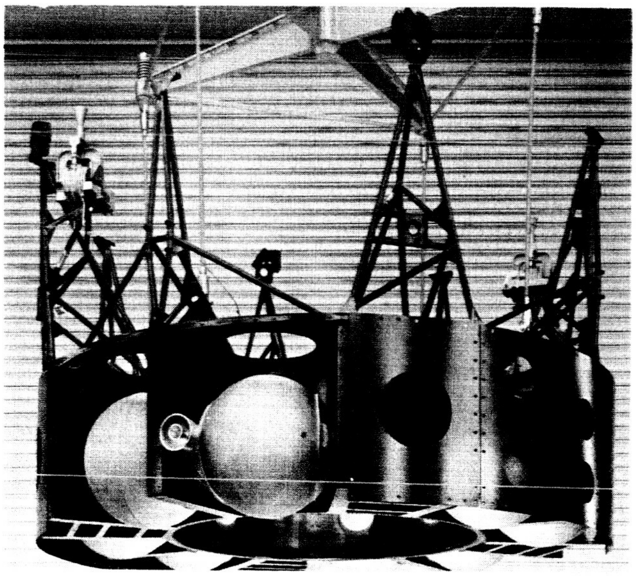
a) Top view of aft assembly



b) Apogee motor brackets

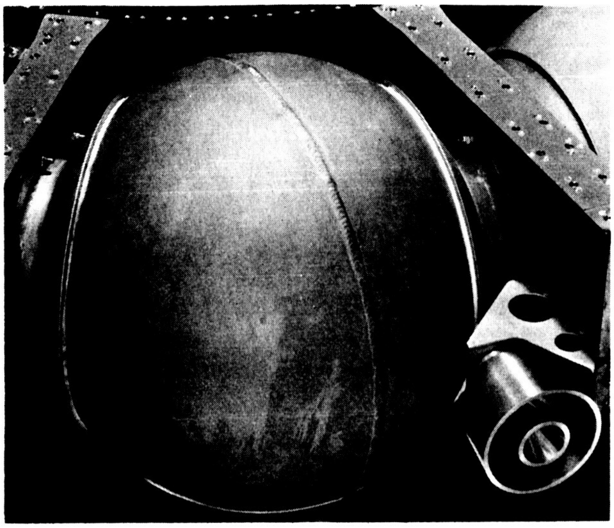


c) Forward assembly

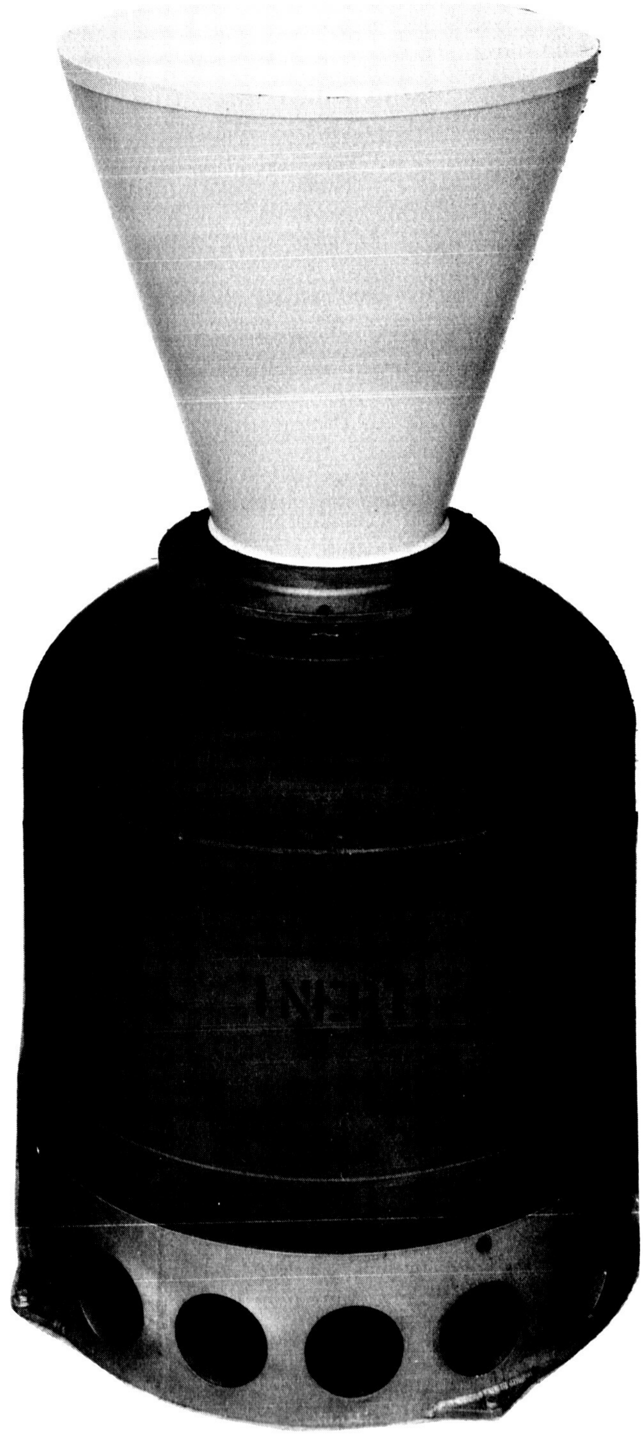


d) Center subassembly

Figure 6-30. Assemblies and Subassemblies of Syncom II

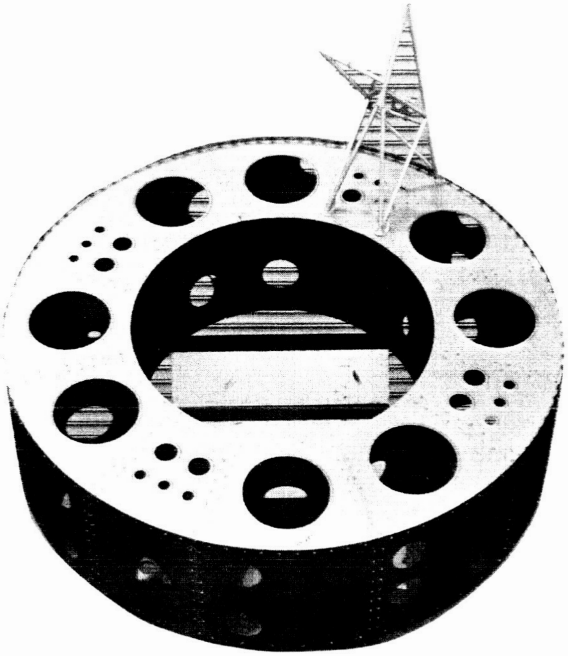


e) Mockup fuel tank and velocity control rocket

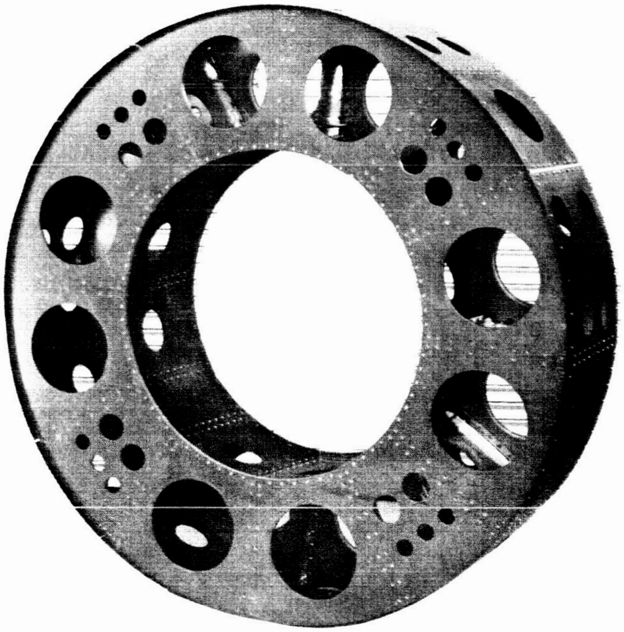


f) Apogee motor mounted on handling and support ring

Figure 6-30 (continued). Assemblies and Subassemblies of Syncom II



g) Top view center subassembly



h) Aft view center subassembly

Figure 6-30 (continued). Assemblies and Subassemblies of Syncom II

Radial brackets will be arranged about the thrust tube to support components in the center structure area. The change in thrust tube diameter requires that a corresponding change be made to the diameter of the Agena interstage interface diameter.

The central assembly consists of an aluminum alloy sheet tube of a diameter slightly larger than that of the aft tube to provide heat insulation about the motor case. Elimination of the radial panels and the outer enclosing structure allows greater access to the equipment in this compartment and reduces the degree of redundancy in the primary structure. The radial panels are replaced with stiffeners attached to the central tube. The bracketry supporting the components in this area will attach to the tube stiffeners or to the radial brackets in the aft structure.

The concept of the solar panel construction and mounting is being investigated to optimize this design. Presently the most promising arrangement is the substitution of a three-point nonconstrained attachment of the panel to the structure in place of the former method of attaching the panel in a small central area. The distributed attach reaction reduces the required bonding stiffness of the panels to assure a satisfactory resonant frequency.

Weight and Balance Analysis

Weight and center-of-gravity data for the engineering model HSX 302-T1 were measured prior to vibration testing. A description of the measuring procedure and results for the actual weight and center of gravity of T-1 are included in this report. Differences from the previous weight statement, submitted in the Summary Report, have been solved to reflect the test results. Table 6-34 shows that 92.9 percent of the vehicle components were actually weighed, 6.4 percent calculated, and 0.7 percent estimated prior to obtaining the actual weight of T-1 in a fully loaded condition.

The engineering model (T-1) was not ballasted to meet the current maximum spacecraft weight at separation from the Atlas-Agena D booster (1518 pounds). It was believed that a loading deviation of approximately 1 percent could be tolerated due to the many expected structural and control system design changes. Consequently, the measurement tests were conducted before the spacecraft was fully instrumented for vibration testing and without the nutation dampers installed because of interference with the battery ballast.

Table 6-35 summarizes the latest weight data for the Suncom II in the planned launch configuration. The current statement includes the first estimate from JPL on moment-of-inertia data for the apogee motor and the latest estimate from Marquardt on weight data for the reaction control system. Structural design changes are in progress but their anticipated weight changes are not included in this report; however, continued updating of this statement will be provided.

TABLE 6-34. SYNCOM II ACTUAL WEIGHT STATUS

Engineering Model HSX 302-T1

Component	*	Weight, pounds
Electronics		(134.62)
Electronics quadrants	A	79.71
Telemeter transmitter	A	2.48
Traveling-wave tube	A	8.84
Telemeter monitor	A	0.17
RF power switch	A	1.08
Power supply, traveling-wave tube	A	10.04
Antenna electronics	C	30.70
Installation hardware electronics	C	1.60
Wire harness subsystem		(11.00)
Wire harness dummy	A	11.00
Power supply subsystem		(107.14)
Battery and regulator, forward	A	31.20
Battery and regulator, aft	A	31.20
Solar cell	A	41.74
Solar panel retainer	A	2.50
Installation hardware, solar panel	C	0.50
Control subsystem		(32.49)
Control spin speed	A	5.00
Tank assembly dummy	A	25.79
Velocity jet dummy	A	0.70
Manifold dummy	C	0.17
Bracket velocity jet	C	0.24
Hardware	C	0.59
Valves	E	-0
Transducers	E	-0
Tubing	E	-0
Thermal shield	E	-0
Miscellaneous fitting and hardware	E	-0

TABLE 6-34 (continued)

Component	*	Weight, pounds
Structure subsystem		(136.81)
Thrust tube	A	11.75
Ring thrust tube	A	5.73
Ring stiffener	A	2.22
Stringer tube	A	7.92
Ribs	A	16.80
Plate panel attachment	A	0.38
Fitting panel attachment	A	1.80
Ring, aft	A	3.18
Bulkhead, aft	A	6.30
Motor mount pad	C	5.40
Hardware	A	0.60
Panel assembly bottle	A	25.44
Outer ring (large)	A	5.25
Outer ring (small)	A	1.75
Ring, inner	A	4.25
Support electronics package	A	0.94
Support electronics package	A	0.32
Hardware	A	1.30
Truss jet	A	2.18
Truss sun sensor	A	1.78
Truss solar panel	A	2.96
Bulkhead, forward	A	5.10
Tee-panel attachment	A	2.40
Support electronics package	A	0.94
Support electronics package	A	0.32
Hoist fitting	A	1.56
Bracket flight timer	A	0.03
Paint dummy	E	1.00
Hardware and miscellaneous	A	1.00
Hardware and miscellaneous	A	3.50
Battery installation, forward	C	5.55
Battery installation, aft	C	7.16
Miscellaneous subsystem		(1.88)
Sun sensor dummy	C	0.80
Timer flight dummy	C	0.20
Nutation damper dummy	E	-0
Thermal switch	E	-0
Pyrotechnic switch	A	0.38
Installation hardware	C	0.50

TABLE 6-34 (continued)

Component	*	Weight, pounds
Ballast subsystem		(57.96)
Ballast installation	C	14.30
Ballast installation	C	8.48
Ballast installation	C	20.66
Ballast installation	C	5.52
Ballast installation, miscellaneous	E	9.00

		W	z-z	Izz	Ixx	R/P
T-1, no motor no fuel		(481.90)	21.66	51.99	37.67	1.38
Fuel and N2, dummy	A	55.00				
Oxidizer and N2, dummy	A	86.50				
T-1 less motor		(623.40)	22.08	65.89	44.71	1.47
Motor case dummy	A	54.00				
Nozzle assembly dummy	A	48.10				
Assembly hardware	A	0.20				
Propellant dummy	A	775.60				
Installation hardware	C	0.40				
T-1 fully loaded		(1501.70)	25.34	84.67	72.08	1.17

*Weight:

		Percent/100:
Actual (A) =	1395.33	0.929
Calculated (C) =	96.37	0.064
Estimated (E) =	10.00	0.007

TABLE 6-35. SYNCOM II ESTIMATED WEIGHT STATUS

Planned Launch Configuration

Subsystem	Weight, pounds	ϕ *	θ **
Electronics	134.7	0.216	0.089
Wire harness	19.9	0.032	0.013
Power supply	108.0	0.169	0.069
Controls, inert	49.4	0.077	0.032
Propulsion, inert	122.2	0.196	0.081
Structure	138.3	0.222	0.091
Miscellaneous	52.9	0.089	0.036

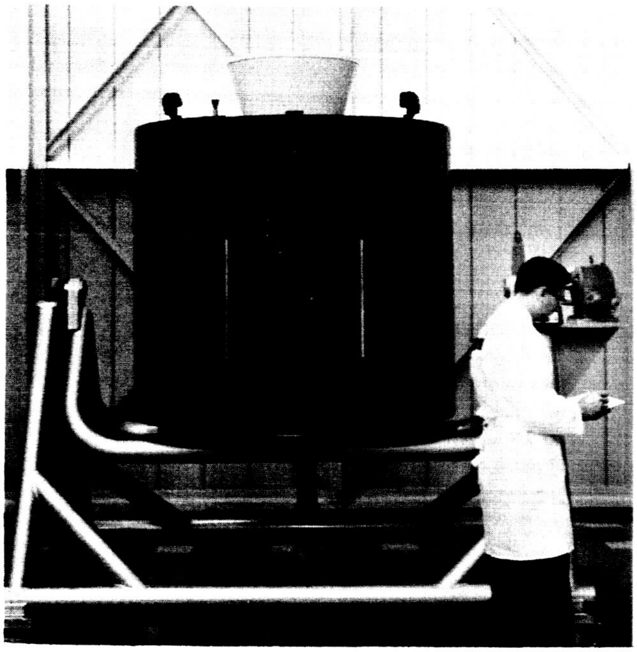
Items	Weight, pounds	z-z	Iz-z	Ix-x	R/P
Final orbit condition	625.4	23.5	56.1	47.2	1.19
N_2 pressurization	2.9				
N_2H_3 -- CH_3 fuel	53.1				
N_2O_4 oxidizer	84.3				
Total at apogee burnout	765.7	23.5	70.4	54.3	1.30
Apogee motor propellant	752.3				
Total payload at separation	1518.0	24.6	87.3	71.5	1.22

*Ratio of subsystem weight to final orbit condition weight.
**Ratio of subsystem weight to total payload at separation.

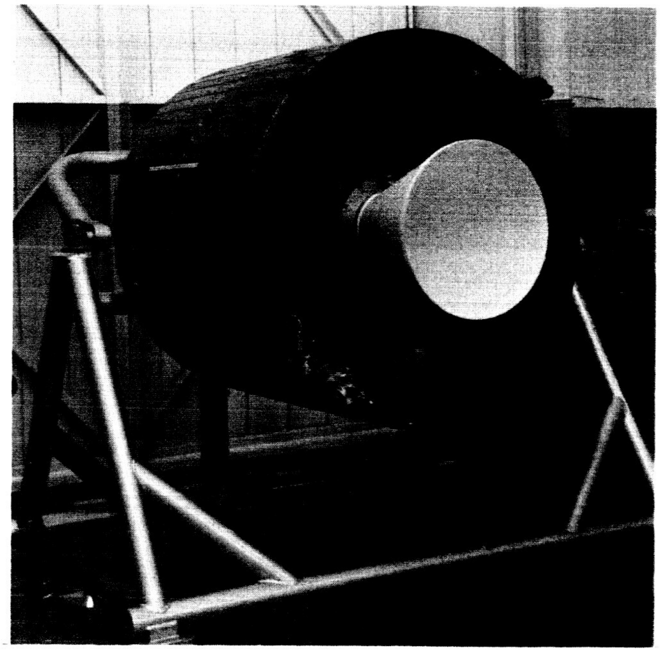
T-1 Weight and Center-of-Gravity Measurement

The purpose of this test was to determine the weight and center of gravity along three axes of the Syncom II HSX-302 T-1 spacecraft and its JPL inert apogee motor. Results of this test are shown in Table 6-36.

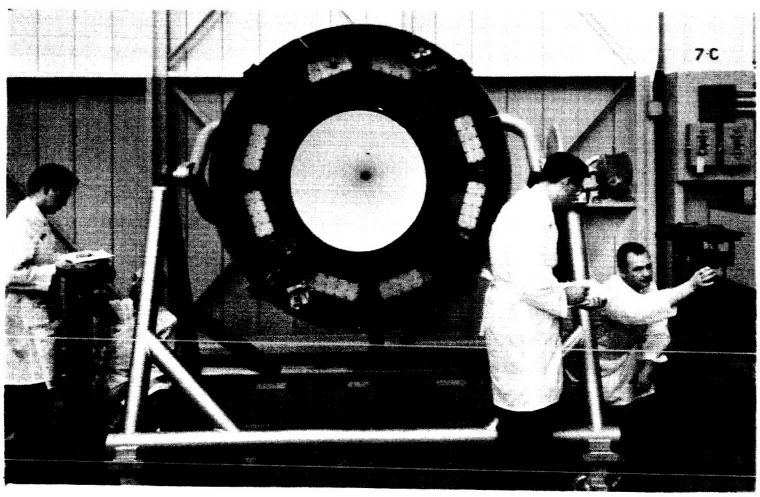
The Syncom II mobile assembly fixture was first leveled while supported on three platform scales. The weight of the fixture was recorded as the sum of the three reactions. Then the assembled spacecraft was attached to the fixture and the reactions recorded; the difference between the two weighings gives the reactions due to the spacecraft. Weighings were accomplished with the spacecraft longitudinal axis in the horizontal position; after the entire spacecraft was rotated 90 degrees CCW about its spin axis, additional scale readings were taken with the longitudinal axis both in the vertical and horizontal positions. Figure 6-31 shows the weight and center-of-gravity measurements in progress.



a) Vertical position



b) Quarter view, horizontal position



c) End view, horizontal position

Figure 6-31. Actual Weight and Center-of-Gravity Test

In order to determine the lateral center of gravity of the JPL apogee motor, the assembled spacecraft was placed on the fixture so that the motor nozzle was up and the reactions recorded. After the motor was removed from the spacecraft another set of scale reactions were recorded. The difference in weight between the two sets of readings gives the reaction due to the motor alone.

In order to determine the longitudinal center of gravity of the apogee motor, the motor was supported by angle extrusions located on two platform scales. This set of readings less the tare readings for the extrusions gives the motor reactions. A weighing was also accomplished with the nozzle and bolts removed so that the nozzle center of gravity could be used in the detailed T-1 weight report.

TABLE 6-36. T-1 WEIGHT AND CG MEASUREMENTS

	Weight, pounds	Z, inches	X, inches	Y, inches
Assembled spacecraft	1501.7	25.34	0.21	-0.33
Inert apogee motor	877.9	27.66	0.125	-0.03
Motor less nozzle and bolts		26.14		
Motor nozzle and bolts		48.79		

Wiring Harness Interface Investigation

The interconnecting wiring harness for Syncom II will be made of only those materials that have been tested and found suitable for space use. Sublimation in a hard vacuum could possibly have an adverse effect on the solar cells. Consequently, only those materials that are usable in their present state or after precleaning in a vacuum will be acceptable.

Optimum effort will be made to hold the harness weight to a minimum through the use of lightweight wire, terminal boards, and connectors, but not to the extent of reducing reliability. The configuration of the harness is determined by the spacecraft structure. Basically the harness will be a full circle near the inner circumference of the structure, with one main breakout to the electronic packages in each quadrant plus minor breakouts as required to solar panels, batteries, sun sensors, terminal boards, any other that may be required. The harness will be laced with unwaxed nylon lacing tape.

Connectors will be provided at each quadrant electronic package and also on the main breakout to each quadrant if so dictated by the design of the spacecraft structure.

The harness will be designed and located to provide the maximum amount of protection from physical damage and flexure, and also to minimize the amount of flexing during installation and replacement of units.

Fabrication of the harness will be in compliance with NASA's specification MSFCPROC-158B, "Procedure for Soldering of Electrical Connectors" (dated 15 February 1963) except in those instances for which deviations have been asked and written approval granted prior to commencement of fabrication.

The harness will be so installed that it is protected from physical damage during the installation or replacement of any spacecraft component, or from excessive heat during or after firing of the apogee motor.

All portions of the harness will be restrained so that damage will not occur from centrifugal force during or after spinup, or from reversal of thrust along the spin axis during boost and apogee motor firing. One method of installation under consideration is securing and supporting the harness assembly within a cover constructed of aluminum, magnesium, fiberglass or other lightweight material of sufficient strength and rigidity. Inside this cover or trough, the harness would be secured by a foam padding material, clamps, or other practical means. The trough must be mounted in the spacecraft so that any section is removable and replaceable with optimum access to the harness cabling. The routing would be designed so that the location will provide only a minimum of interference with the units and function of the spacecraft, and it will be possible to replace or repair the individual electronic packages without removing the entire harness assembly.

Another concept under consideration is the design presently used on the Syncom I spacecraft, which consists of the use of teflon-lined metal clamps. The location of these clamps would avoid critical stress points along the harness. At points at which no installation of mechanical bracketry is possible because of space limitations and insufficient structure to support the bracketry, the harness would be secured directly to the respective equipment units with unwaxed nylon tape.

A third design concept under consideration involves the use of some form of channel hardware with the wiring harness retained and supported within the hardware with potting compound. This would then become an integrated unit, which could be installed rapidly and safely. The wiring would be completely checked out for continuity, hi-pot, etc., after potting and prior to installation.

Other proposed methods of installation are being investigated and the most feasible will be adopted following finalization of the basic structure configuration.

Dynamic Response Survey Data (T-1 Model)

The vibration tests currently being conducted are providing strain and acceleration data for structural design and for unit qualification, in addition to demonstrating the adequacy of the structure in the qualification test environment. The test program is being conducted according to the sequence indicated in Table 6-37.

The qualification test environment has been defined by NASA* (Table 6-37) and the sinusoidal portion of this environment is given below for reference. The following levels are applied along three axes in logarithmic sweeps at 2 octaves per minute, 4.35 minutes duration per axis.

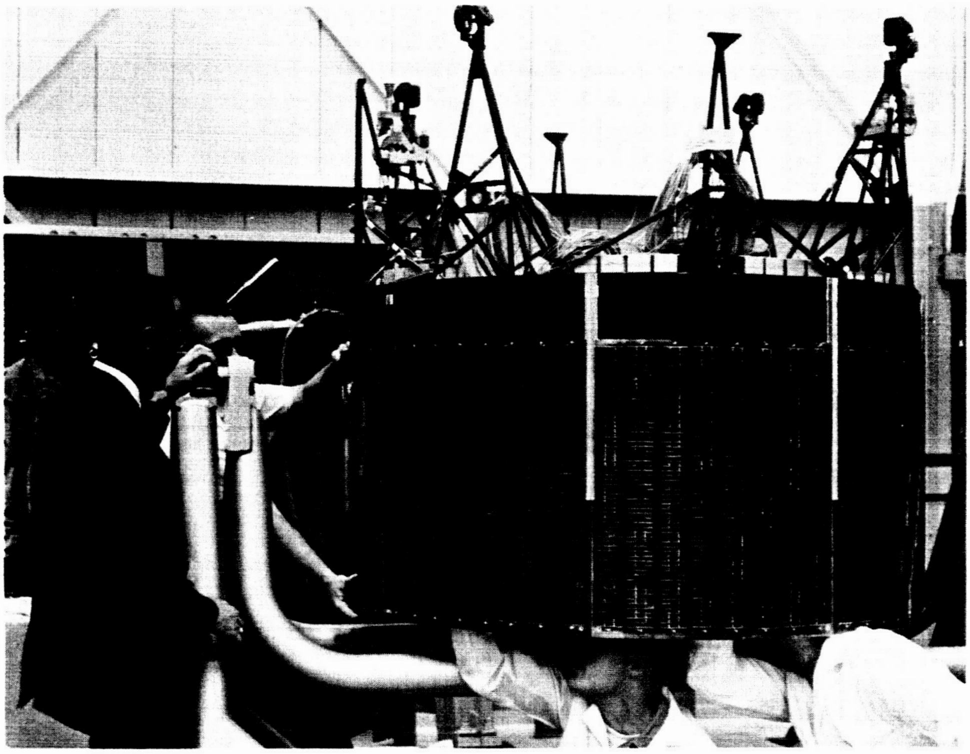
Frequency, cps	Level
5 to 15	0.25 inch double amplitude
15 to 250	3.0 g peak
250 to 400	5.0 g peak
400 to 2000	7.5 g peak

NASA has stated that an exception to these levels may be taken when the predominant longitudinal and lateral frequencies are sufficiently decoupled from those of the Atlas/Agena with the spacecraft attached. In this case, the spacecraft response at the center of gravity may be limited to that of the separation plane input in the range of the predominant spacecraft lateral and longitudinal frequencies for the sinusoidal excitation only. The sinusoidal qualification test environment for input at the apogee motor attachment has been verbally established by NASA to be the same as that applied to the Agena interface.

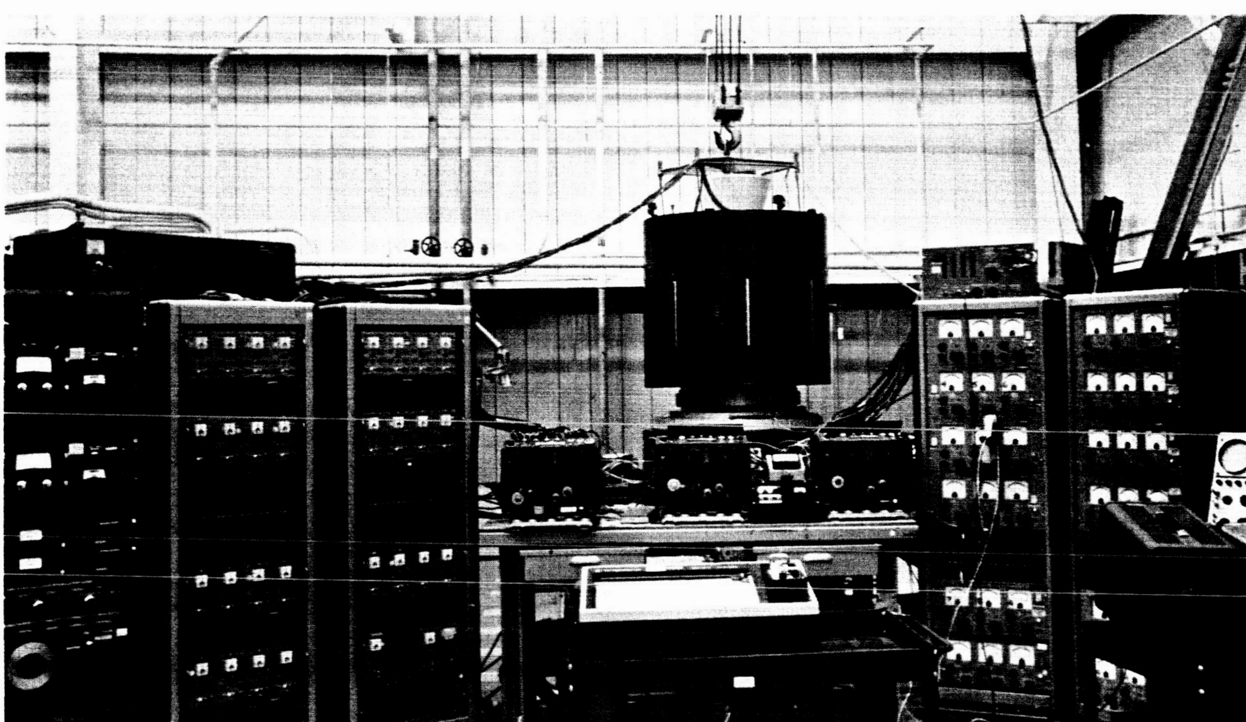
The T-1 spacecraft represents the 1518-pound version of Syncom II. The structural elements conform to present flight hardware design, and the major portion of electronic and control system components are modeled by rigid masses. The solar panels are of current structural design and have simulated solar cells attached. The measured weight and center of gravity, obtained just prior to testing, are 1501.7 pounds and Station 25.34, respectively. The center of gravity is measured from the Agena interface. The measured center of gravity has a radial misalignment of 0.392 inch.

During testing the spacecraft is connected to an overhead crane through eight nylon safety lines, which are attached to the apogee motor and to the spacecraft hoist fittings. The lines are each capable of supporting 4000 pounds and are kept slack during testing. Spacecraft final assembly and installation on the shaker are shown in Figure 6-32.

*"Syncom II Engineering Model Vibration Environment," TWX from A. E. Jones, GFSC, to P. E. Norsell/R. A. Browne.



a) Assembly



b) General arrangement

Figure 6-32. T-1 Spacecraft Vibration Testing

The spacecraft is attached to the Ling 249 shaker or the Team hydrostatic slide table by rigid fixtures (Figure 6-33a). Figure 6-33b shows an operational test. Figure 6-34 gives the excitation axes and component numbering system.

The spacecraft instrumentation consists of 35 crystal accelerometers and 28 strain gauge channels. The accelerometers are mounted on small phenolic blocks and are relocated between runs to obtain the required responses. The distribution of accelerometer blocks is shown in Figure 6-34. The strain gauges are distributed over the thrust tube, ribs, tank mounting panels, and trusswork to provide strain data in these locations. Strain gauge locations are shown in Figure 6-35. A block diagram of the data acquisition and reduction systems is presented in Figure 6-36.

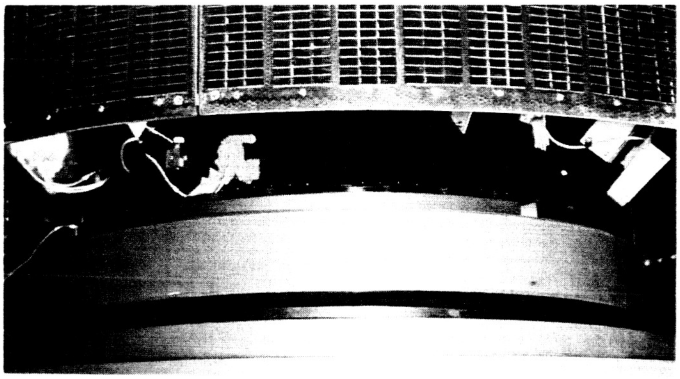
The testing completed thus far includes all of tests 1 and 2 (Table 6-37). Representative plots of phase angle versus frequency and amplification factor versus frequency are presented in Figures 6-37 through 6-41 for the qualification test levels. These results are preliminary and will be finalized upon detailed examination of the test data. It is concluded from these preliminary results that the predominant longitudinal frequency is 123 cps. Phase angle is defined for these plots as $\tan^{-1} \frac{\dot{x}_{output}}{\ddot{x}_{input}}$ and amplification factor as $\frac{\ddot{x}_{output}}{\ddot{x}_{input}}$. Investigations were conducted during test 1 to determine structural damping at the lower resonant frequencies and to establish any changes in resonant frequencies due to sweep direction. These data are being examined and will be reported subsequently. The frequency sweeps in test 1 were performed at 1 octave per minute to reduce the effect of sweep rate on response.

Difficulty was encountered during tests 1 and 2 in maintaining proper input levels to the structure. The shaker and control units that contributed to this condition are being isolated during the shaker investigation and will be corrected before proceeding with test 3.

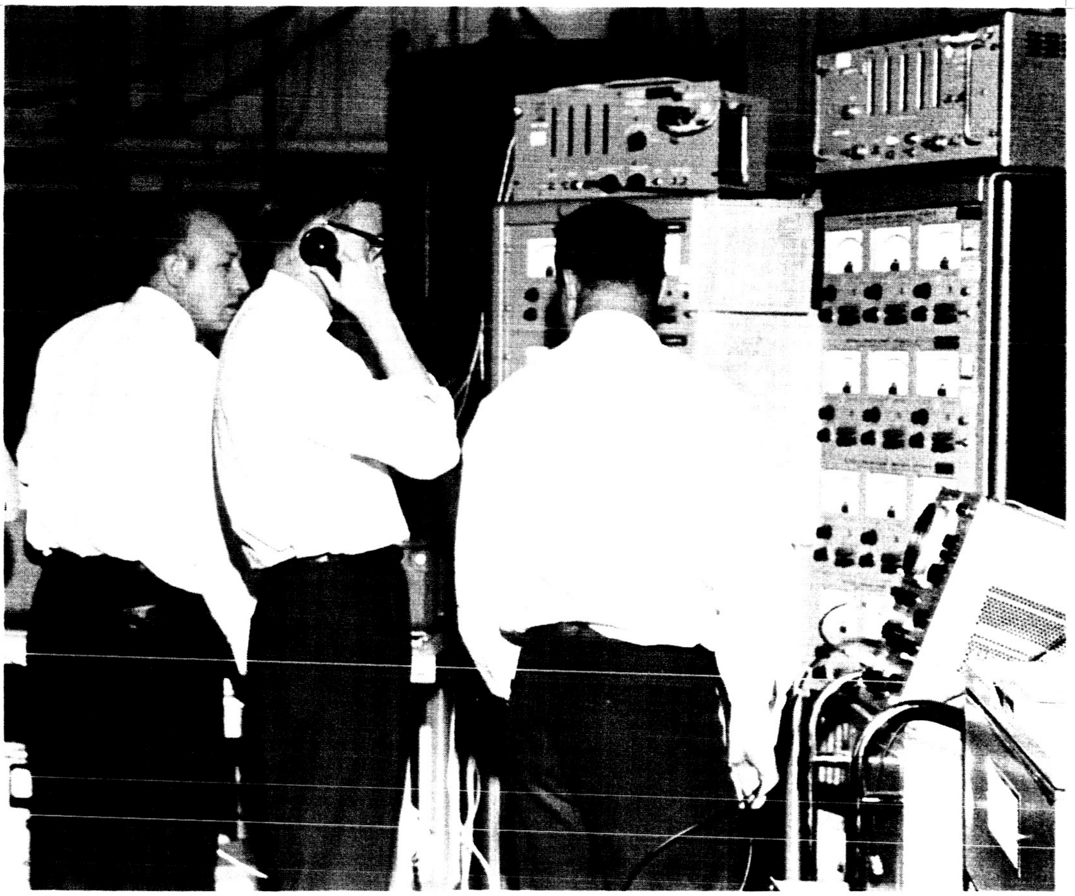
The spacecraft was disassembled following test 2 for relocation of accelerometers and inspection of the structure. The following conditions were observed following the thrust direction qualification test:

1) Some of the dummy battery cells located between ribs 2 and 3 (Figure 6-35) and between ribs 6 and 7 had slipped partially out of the packages and were resting against the outer ring.

2) The torque on one of the two apogee motor mounting bolts at rib 11 was 60 in-lb instead of the required 90 in-lb. This bolt was inspected and found to have battered threads over approximately 30 percent of the thread length.



a) Separation interface fixture



b) Activity

Figure 6-33. Vibration Test, T-1 Spacecraft

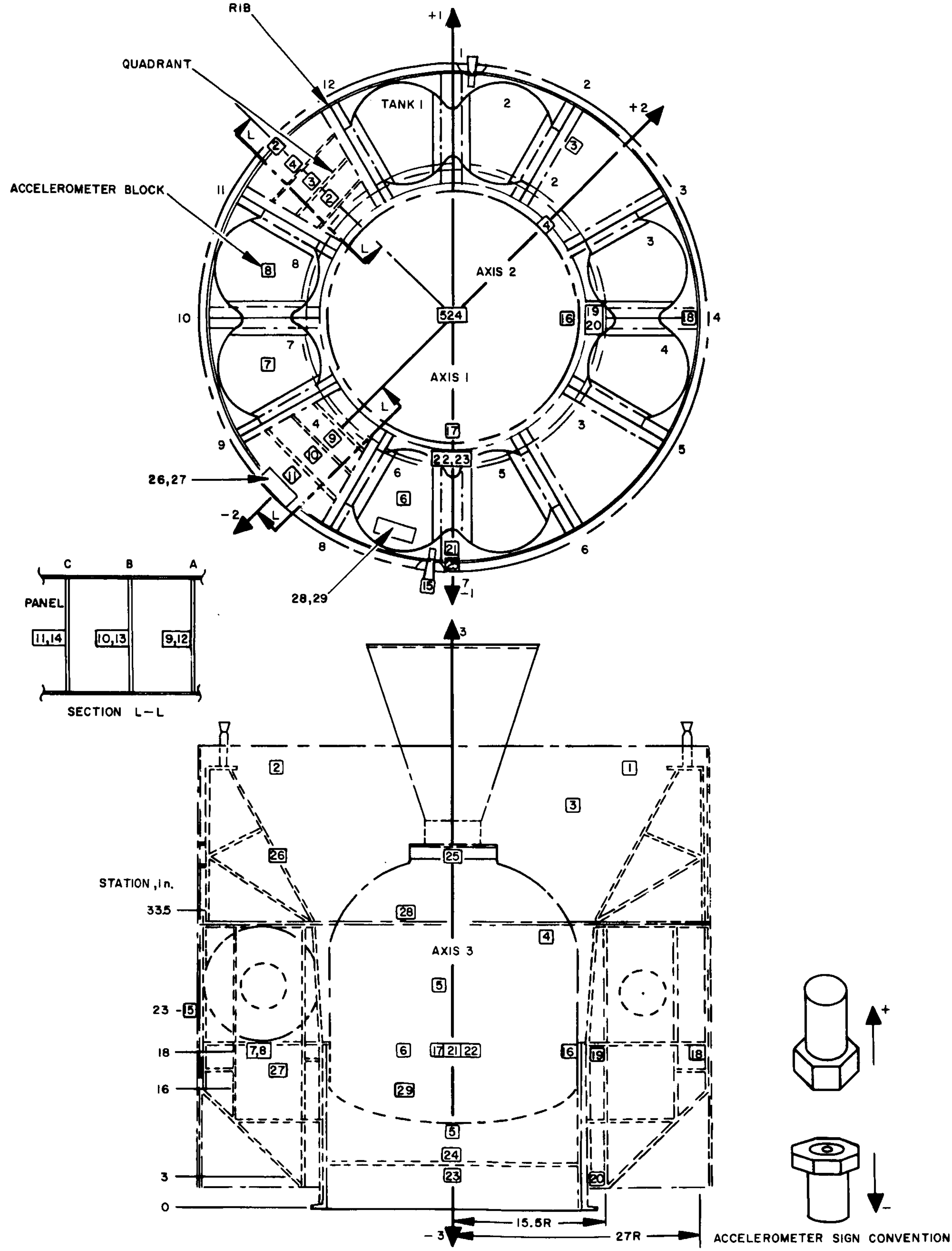


Figure 6-34. Excitation Axes and Accelerometer Locations

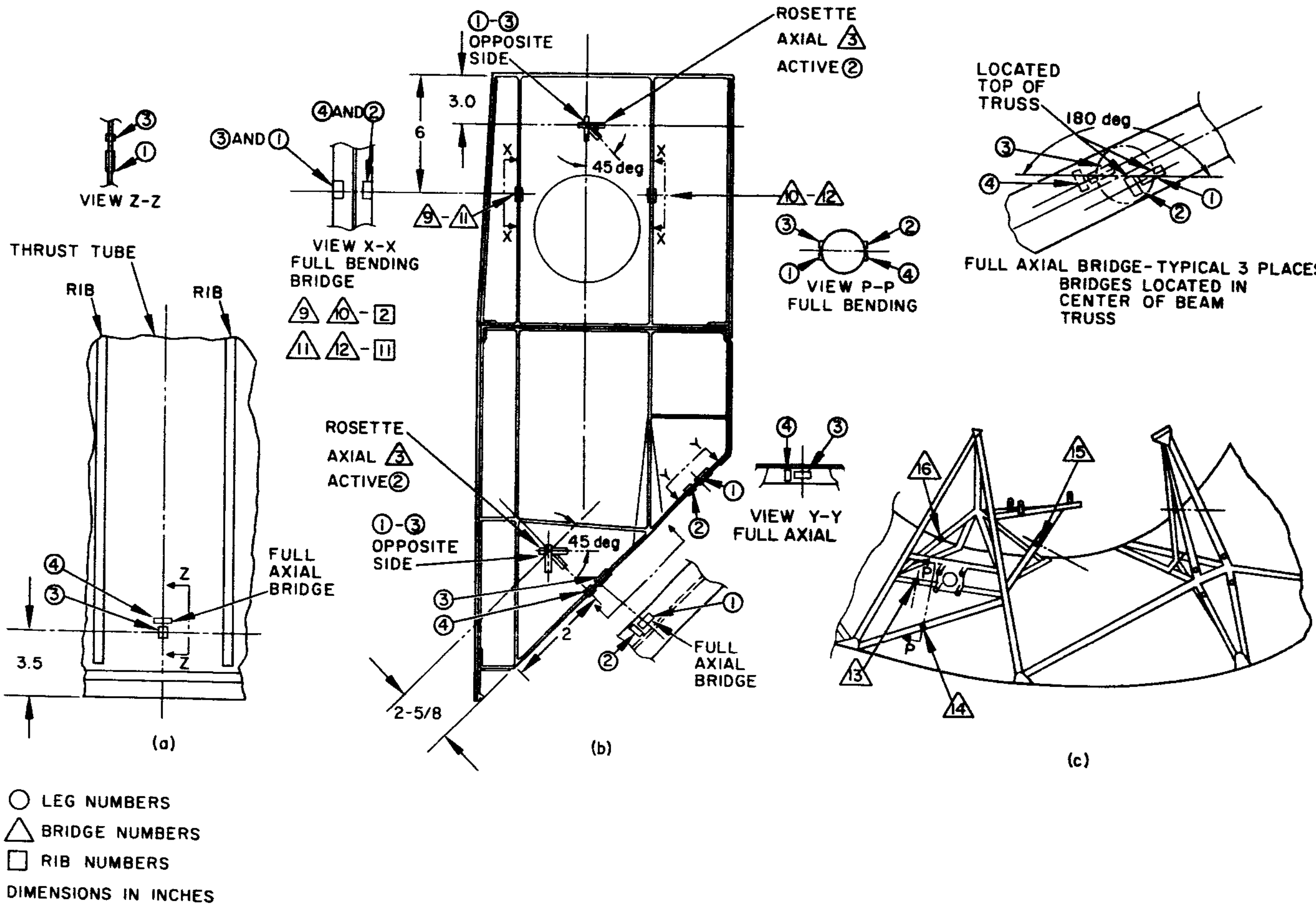


Figure 6-35. Strain Transducer Locations

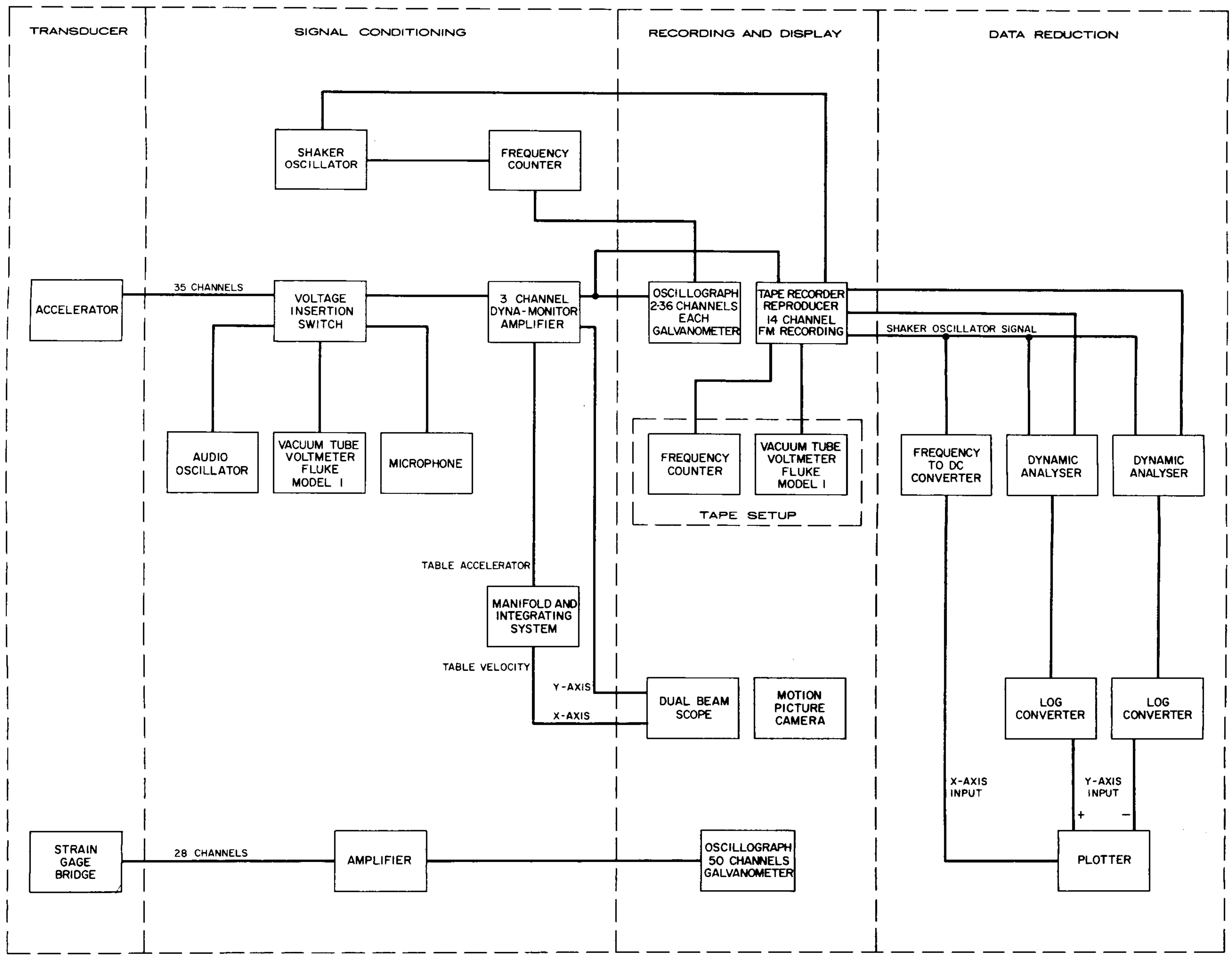
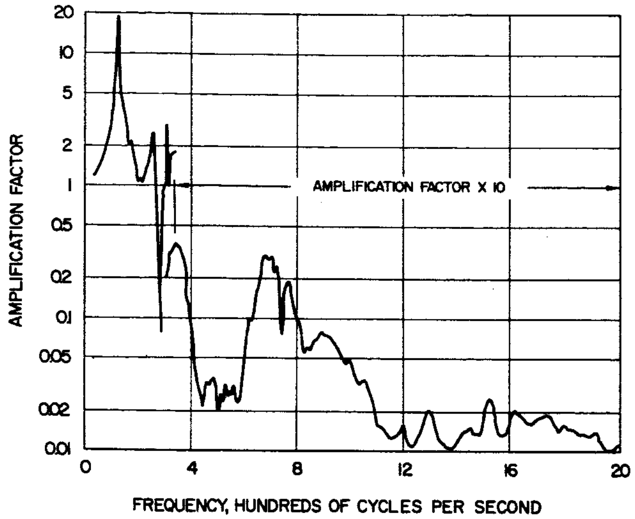
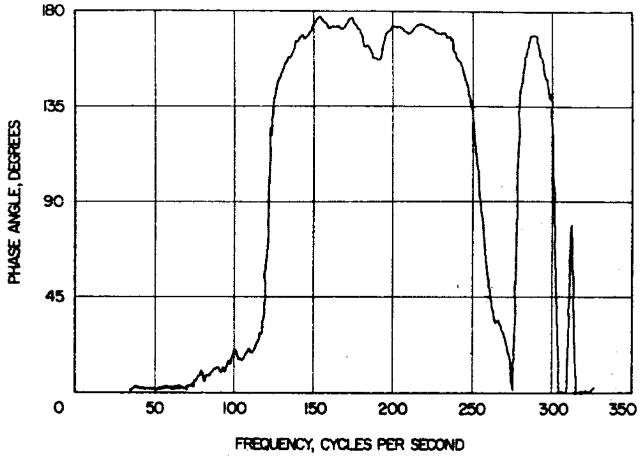


Figure 6-36. Instrumentation Block Diagram, Syncom II T-1 Vibration Test

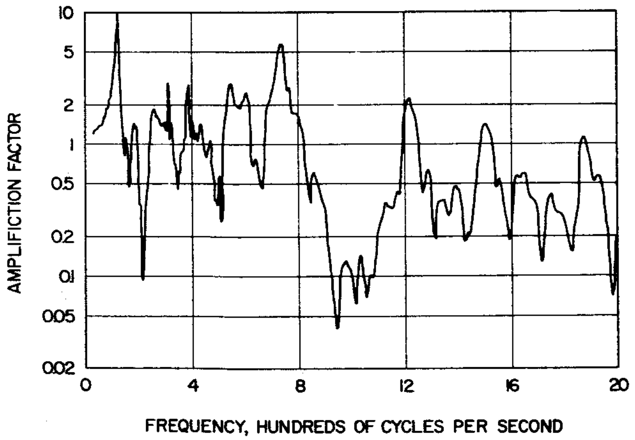


a) Amplification factor

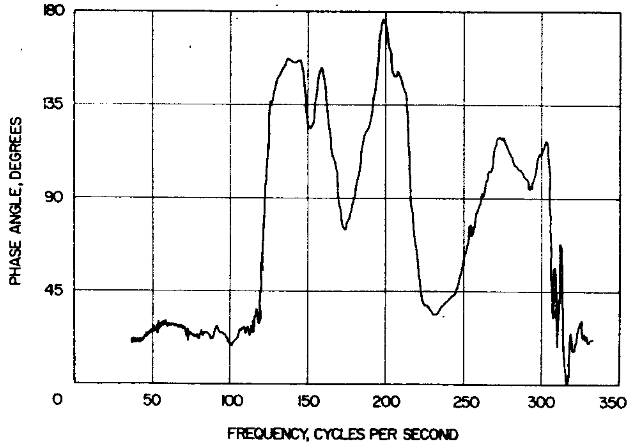


b) HSX-302-T1 vehicle phase angle

Figure 6-37. Top of the Apogee Motor, Axial Direction, Qualification Test Level

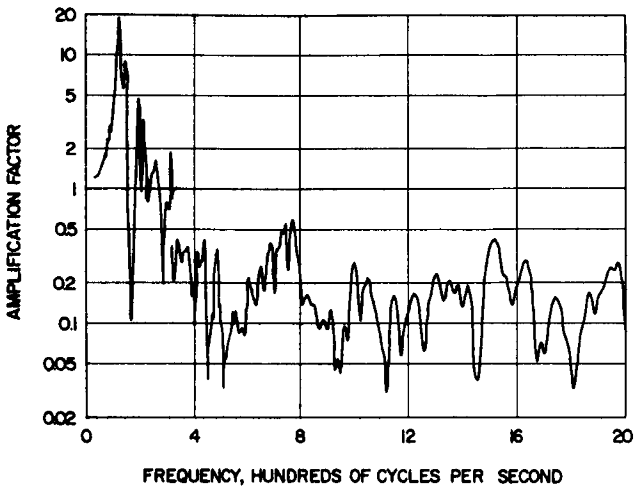


a) Amplification factor

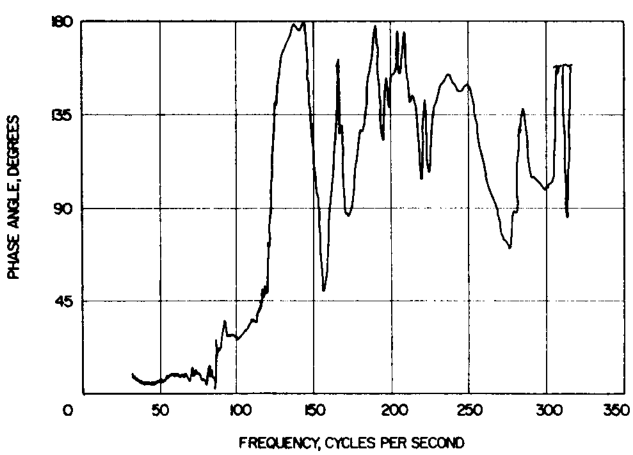


b) HSX-302-T1 vehicle phase angle

Figure 6-38. Bottom of the Bipropellant Tank 7, Axial Direction, Qualification Test Levels

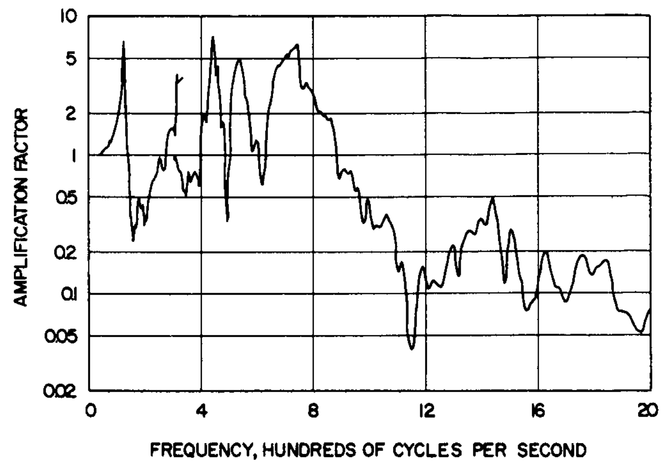


a) Amplification factor

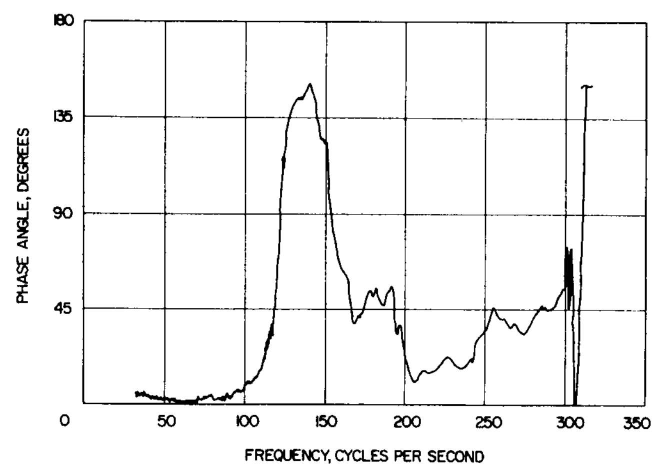


b) HSX-302-T1 vehicle phase angle

Figure 6-39. Quadrant 4 Electronics, Axial Direction, Qualification Test Levels

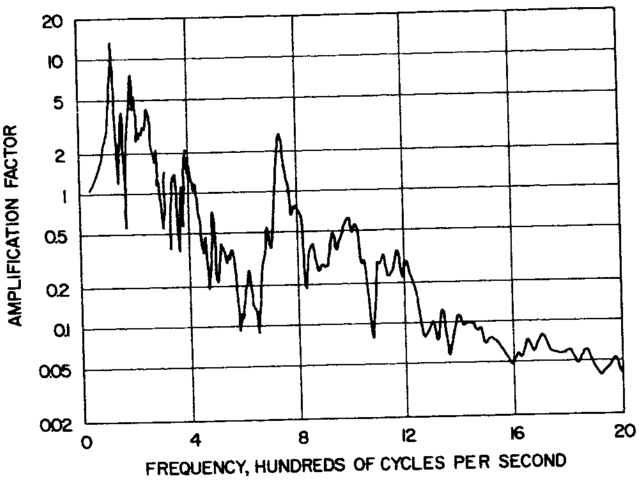


a) Amplification factor

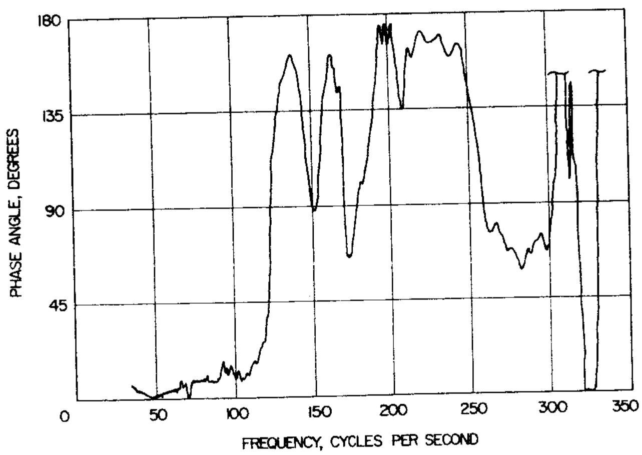


b) HSX-302-T1 vehicle phase angle

Figure 6-40. Apogee Motor Attachment at Rib 8, Axial Direction, Qualification Test Levels



a) Amplification factor



b) HSX-302-T1 vehicle phase angle

Figure 6-41. Battery Pack above Bipropellant Tank 6, Axial Direction, Qualification Test Levels

3) Solar panel 6a sustained a concave face sheet separation along an axial edge for approximately 60 percent of the length. The failure was approximately 1 inch wide, 1 inch in from the edge of the panel, and mainly along the unsupported portion of the panel edge.

4) Solar panel 7a sustained a concave face sheet separation similar to that of 6a.

5) The solar cell covers along one axial edge of panel 4a were crushed from battering against the restraining clips.

The above face sheet separations differ from those experienced during the solar panel tests since: 1) the axial edges of the panels were restrained, and 2) the lateral inputs to the panels were much lower at the predominant panel frequency. Failure of solar panels 6a and 7a can be attributed to the low resin content in the panels at the separated face sheet locations. These panels will be repaired before they are subjected to additional vibration testing.

TABLE 6-37. VIBRATION TEST PROGRAM

Test	Input Location	Excitation Axis	Sinusoidal Input Level
1	Agena interface	Longitudinal	1/2 to 1 g peak
2	Agena interface	Longitudinal	Qualification
-	Shaker investigations		
3	Agena interface	Lateral -1	1/2 to 1 g
4	Agena interface	Lateral -1	Qualification
5	Agena interface	Lateral -2	Qualification
6	Apogee motor attach	Lateral -1	1/2 to 1 g peak
7	Apogee motor attach	Lateral -1	Qualification
8	Apogee motor attach	Lateral -2	Qualification
9	Apogee motor attach	Longitudinal	1/2 to 1 g
10	Apogee motor attach	Longitudinal	Qualification

THERMAL CONTROL

A simple analytical model of a 90-degree sector of the Syncom II has been completed and its basic parameters obtained and several test cases have been completed using the computer.

Parametric studies using this model are now being prepared. These studies will be analyzed and used to determine effects of solar inclination angles, high and low internal power dissipation, degradation of thermal coatings resulting from vacuum, and ultraviolet effects.

APOGEE INJECTION ROCKET MOTOR

The Syncom II apogee engine is similar to the JPL-developed Syncom I apogee engine with respect to configuration, materials, and propellant formulation. The engine will provide a velocity increment of 6100 fps for an injected spacecraft weight of 1518 pounds. An off-loading capability commensurate with a spacecraft weight of 1300 pounds has been incorporated into the design.

Development Program Progress

Heavy-weight engine cases, currently on order, are due during the week of 29 April 1963 at JPL; flight-weight cases are to be delivered during August 1963 for developmental tests. Heavy-weight truncated conical nozzles are on order, and contoured flight nozzles are in the late design phase. The initial heavy-weight test will be conducted during June 1963, and the initial flight-weight test during September of the same year.

To date, three of the four planned subscale tests have been conducted, utilizing Syncom I engine components. The purpose of these tests was to evaluate performance at simulated altitude conditions with conical nozzles and to ensure adequate performance of the new altitude cell at Edwards Air Force Base. The test program has been successful and the altitude simulation (HYPROX) system operated satisfactorily.

Installation of the 150-gallon mixer at Edwards Air Force Base, which is to be used for loading Syncom II engines, is proceeding on schedule. Mixer operation will be turned over to JPL during September 1963.

7. SPACECRAFT RELIABILITY

RELIABILITY FAILURE MODE

Introduction

The inherent reliability of any system is established by the basic design. During the early design stage, reliability can best be improved by thorough analysis of how the hardware can fail and the effects the failure has on the success of accomplishing the intended mission.

Two useful techniques in making reliability decisions are failure mode analysis and failure effects analysis. Both of these reliability tools can be used to actually improve the reliability of the equipment by requiring a systematic review of the design. They both provide classifications of failures. In Syncom II they will be combined as one technique.

Failure mode analysis is concerned with the physics of failures. All conceivable failures are listed; mechanisms which may induce failures are included. The individual failures are classified according to seriousness and probability of occurrence. The technique will be used for both mechanical and electronic parts, but in the case of the latter, care must be exercised to determine the physics of failure. For example, a noisy circuit might involve poor soldering, improperly mounted tube elements, solder particles, and many similar problems often associated with workmanship.

Failure effects analysis is concerned with the effects that a failure may have on the mechanical or electronic configuration to which it is attached. All failures are listed that could prevent the configuration from translating mechanical impulses to other mechanical configurations. In the case of a circuit, the failures listed are those that could prevent transmission of correct signals to other circuits in the system. These failures can then be classified as in the failure mode analysis, but in this case, they are classified by the seriousness of each failure as it affects other parts in the system.

Plan Summary

Failure mode and effects analysis will be performed at all functional levels within the spacecraft, including evaluation of components, units, assemblies, quadrants, subsystems, and the complete system. The evaluation will be implemented by responsible design engineers using a failure mode and effects analysis procedure prepared specifically for Syncom II. Summary tables of analysis will be included as part of the necessary data for scheduled design reviews.

Ten days prior to a scheduled design review, completed failure mode and effects analysis tables on the equipment to be reviewed will be submitted to the project reliability office for consideration and review. Failure mode and effects analysis of system, subsystem, and quadrant equipment will be scheduled and reviewed prior to scheduled assembly, unit, and component reviews. For application to Syncom II the spacecraft has been divided into functional block diagrams shown below.

I	Receiving Antenna Unit Assembly
II	Multiple-Access Transponder Receiver Assembly
III	Frequency Translation Transponder Receiver Assembly
IV	Transmitter Unit Assembly
V	Transmitting Antenna Unit Assembly
VI	Phased-Array Control Electronics Digital Assembly
VII	Phased-Array Control Electronics Analog Assembly
VIII	Sun Sensor Assembly
IX	Reaction Control Unit Assembly
X	Battery and Charging Assembly
XI	Solar Panel Assembly
XII	Structure Assembly
XIII	Apogee Engine
XIV	Wiring Harness
XV	Telemetry Encoder Assembly
XVI	Telemetry Transmitter Assembly
XVII	Telemetry and Command Antenna Assembly
XVIII	Command Decoder Assembly
XIX	Command Receiver Assembly
XX	Nutation Damper
XXI	Apogee Engine Firing Assembly

<u>Note:</u> Regulators are considered part of the assemblies

General Description of Method

The detailed analysis procedure will include a complete list of instructions and forms for completion of the failure mode and effects analysis. The forms are tables prepared showing a list of:

1) Potential failures and malfunctions

2) The probability of occurrence of each failure

3) A prediction of the ultimate effect that each failure would have on successful mission completion.

Failure mode and effects analysis applied at the unit or subsystem level assumes a different form from that used at the part level. At the part level minute impurities, lattice structures and the like are of interest; at the unit level much larger entities and their more complex modes of failure such as shorts, opens, breakage, binding, shearing, etc., are of concern.

Generally, the analysis begins with the preparation of a block diagram of the system to be analyzed by the design engineer followed by the completion of the columns in the failure mode and effects analysis tables. Although the analysis procedure is well defined, competent reliability specialists and senior technical consultants will be available to assist the design engineer when required. An example of a failure mode and effects analysis form is shown in Table 7-1. The various columns found on the form are described below:

Item Failure Description. Particular attention is directed toward providing an accurate and complete description of the particular failure of the complete item (unit or subsystem) for which the analysis is being made.

Part, Component, Unit, or Subsystem. List each element of the item being analyzed to the appropriate level at which the analysis is being made. Symbol designations are useful.

Description. Use part numbers, part values, or common names, as appropriate.

Derating Factor. Applicable to electronic parts as the ratio of applied stress to rated functional stress. For mechanical parts the reciprocal of derating factor, or the safety factor, is commonly used.

Description of Assumed Failure. List all conceivable failures, including both degenerative and catastrophic types. The designer is not to list merely what he thinks will happen, but everything which could possibly fail. Effects of environmental factors and all functional stresses are to be evaluated in relation to their failure including capability. The mechanism and cause of failure should be listed.

Item Failure Mode. State what happens as an integral or related part of the item failure description as a direct result of the element failure.

Influence on Next Element of System. The consequence of the failure on the system performance and/or the mission should be described. Not all failures result in reduction of mission capability. An effectiveness scale is useful here.

$P(f_i)$. The probability of the particular element failure.

$P(F/f_i)$. The probability of item failure, given the element failure. For degradation-type failures and for redundant element failures this probability may have a range of values.

$P(F_i)$. Probability that item fails by this mode.

Basis for Estimate of Probabilities and/or Remarks. In discussion among the designer, reliability specialist, and senior technical consultants, the probability of occurrence of each failure is determined. Effects of probability, time, performance, and environment must be considered.

Possible Methods to Eliminate Failure Mode. Is it technologically possible to eliminate the mechanism or cause of failure? Can the opportunity for failure be removed, e.g., by eliminating or changing the part?

Reasons for Nonobviating Failure Mode. List the reasons why the failure cannot be obviated, any of which, in the design function if incorporated to eliminate failure.

Calculated Reliability. If the total $P(F_i)$ is less than 0.10, a good approximation to the corresponding reliability is $1 - \Sigma\ (F_i)$.

How is Failure Detected? By abnormal operation, telemetry data, etc.

TABLE 7-1. RELIABILITY FAILURE MODE AND EFFECTS ANALYSIS

PROJECT________ MISSION PHASE________ DATE________
DRAWING________ ITEM NAME________ SHEET____ OF____
REQUIRED RELIABILITY________ CALCULATED REL.________ BY________
ITEM FAILURE DESCRIPTION________

PART, COMPONENT, UNIT, OR SUBSYSTEM	DESCRIPTION	DERATING FACTOR	DESCRIPTION OF ASSUMED FAILURE	ITEM FAILURE MODE	INFLUENCE ON NEXT ELEMENT OF SYSTEM	$P(f_i)$	$P(F/f_i)$	$P(F_i)$	BASIS FOR ESTIMATE OF PROBABILITIES AND/OR REMARKS	POSSIBLE METHODS TO ELIMINATE FAILURE MODE	REASONS FOR NONOBVIATING FAILURE MODE	HOW IS FAILURE DETECTED?
							TOTAL		CALCULATED $R \approx 1 - \sum P(F_i)$ $R \approx$			

$P(f_i)$ = PROBABILITY OF PART FAILURE

$P(F/f_i)$ = PROBABILITY OF ITEM FAILURE IF PART FAILURE OCCURS

$R(F_i)$ = PROBABILITY OF ITEM FAILURE

SYSTEM AVAILABILITY AND RELIABILITY STUDIES

The Syncom II system availability studies have included during this reporting period a formal presentation to NASA, Goddard Spaceflight Center, of the work performed by Hughes and a continued effort to develop mathematical models for an optimum replacement policy.

Probabilistic models developed and presented up to this time have been based on one operable spacecraft in orbit with an old-age replacement policy. The present effort involves the extension of this work to encompass techniques which take advantage of the remaining useful lifetime of an operable satellite that has been replaced. In addition, the computer program developed for simulation of the spacecraft reliability model is being extended to compute communication channel transition probabilities for inclusion in a redefined availability model. These probabilities will be used to determine the probable spacecraft configurations and optimum replacement policy based on the state of health of the spacecraft during orbit. The results will be presented in a later report.

The reliability studies have included an examination of the hypothesis that four individual Syncom spacecraft with only one quadrant of electronics each would provide a higher probability of communication mission success for the long lifetime orbital requirement than a single four-quadrant Syncom II. This hypothesis has been tested and rejected by a detailed evaluation of the general reliability function R(t) for one of four and four of four transponder quadrants of the Syncom II configuration, including common equipment, versus one of four and four of four separate spacecraft. The results of the evaluation are shown in Figure 7-1 . The configuration of the four-quadrant Syncom II is shown in Figure 7-2 . Each separate spacecraft may be characterized by those equipments shown in one quadrant of this figure. (The batteries and charging circuitry are only twofold redundant for the separate spacecraft configuration.) This comparison confirms the desirability of the Syncom II concept of a single multichannel spacecraft and further illustrates the superior capability of one spacecraft for total communication performance.

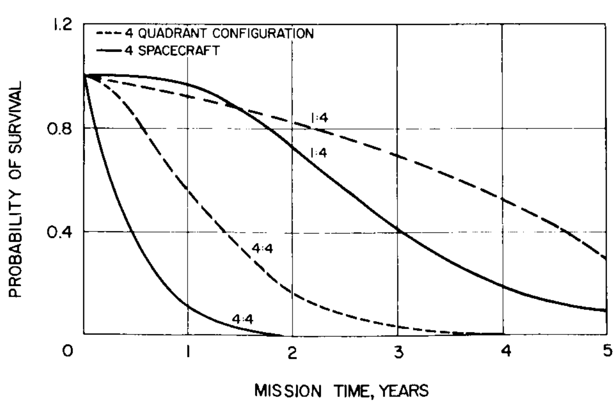


Figure 7-1. Comparison of Four Quadrants versus Four Spacecraft

Multiple access communications

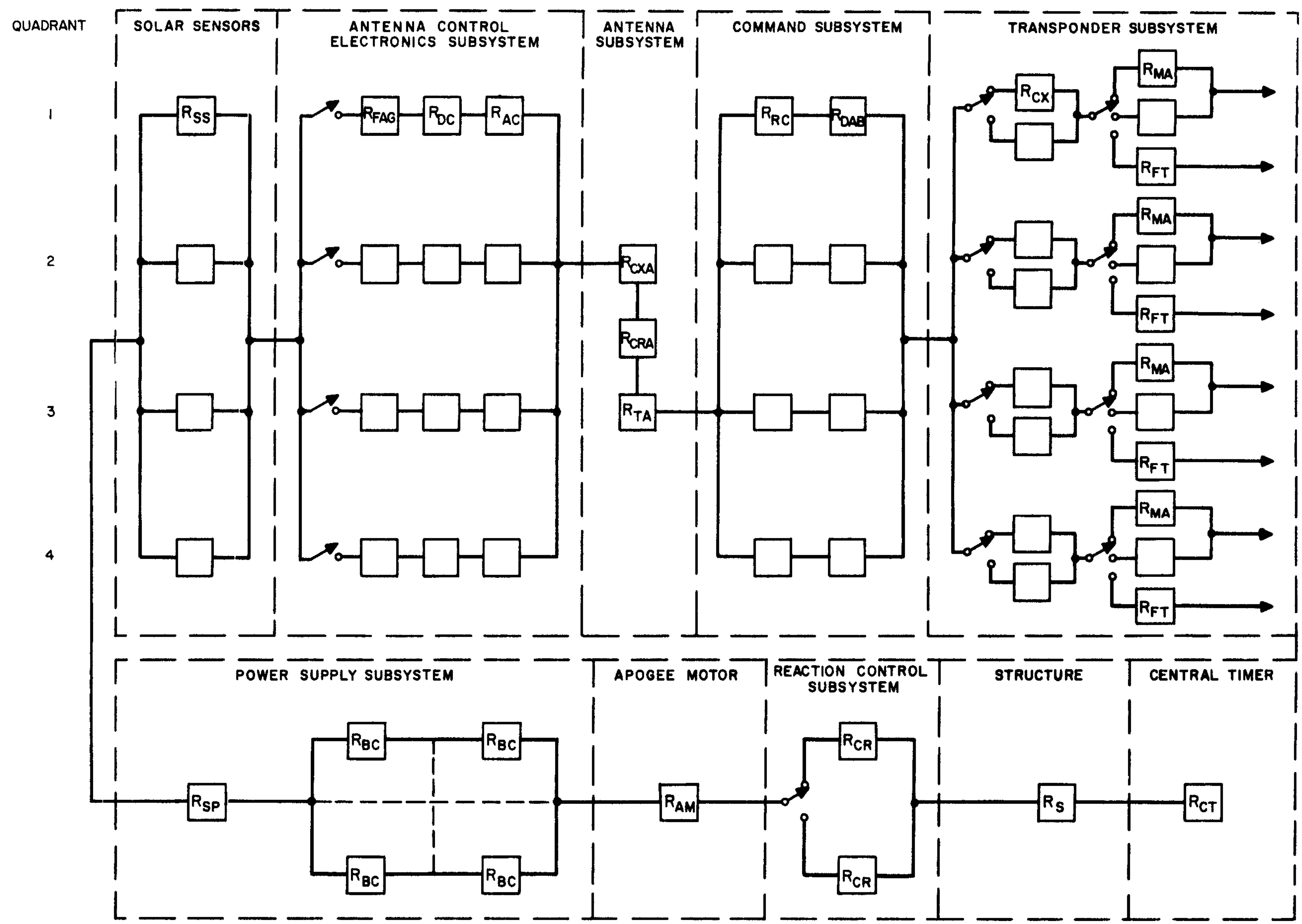


Figure 7-2. Communication Reliability Diagram - Syncom II

CRITICAL COMPONENT TEST PLANS FOR SYNCOM

There are several components to be used in the Syncom II spacecraft that will be given special emphasis in test and selection: 1) separation switches, 2) batteries, 3) solar sensors, 4) central timer, and 5) traveling-wave tubes.

The detailed specifications for each of these components will include specific acceptance criteria to ensure conformity to the design, processes, fabrication, and environmental requirements. Acceptance tests will be performed on a 100-percent basis. Qualification and the special testing will be performed on a selected sample of these components. The special testing designated in the following test plans is in addition to the normal development, qualification, and acceptance testing of these components.

Reliability management of the special component program will include participation in the following:

1) Preparation of detailed design and test specifications.

2) Vendor selection and monitoring.

3) Qualification and special test monitoring.

4) Failure monitoring, analysis, and corrective action.

5) Evaluation of test data.

6) Implementation of controls and procedures (design reviews, scheduling, etc.).

The special test plan for each of the above components is described on the following pages.

Separation Switches

Introduction

It is the purpose of this test plan to evaluate and demonstrate that the separation switches will perform adequately during the Syncom II mission profile.

The separation switches will be used to 1) short the apogee engine firing circuit until spacecraft separation from the Agena booster, and 2) to provide dc power to the central timer at the time the spacecraft separates from the Agena booster.

The exact number of switches or the specific type has not been determined at this time.

Applicable Documents

1) Switch Procurement Specification

2) Switch Qualification and Acceptance Test Plan

General Requirements

The test plan is based on the assumptions that the nature of operation of the switch is "one-shot" and that the environments described are representative of mission conditions. Testing will be accomplished to 1) show the ability of the switch to operate under expected mission conditions, and 2) demonstrate a reliability of 0.999 with a reasonable confidence.

The switches used for this test will be from the same lot as those used in flight spacecraft.

Test Program

Twelve switches will be selected for the following program. The environments that the switches will be subjected to are as follows:

1) Vibration (combined sinusoidal, random and shock)

2) Acceleration (boost and spin)

3) Acoustical noise

4) Thermal-vacuum (12 hours with one cycle of highest and lowest expected temperature)

5) Low-temperature operation

6) High-temperature operation

The environmental levels and durations will be in accordance with the Qualification Test Specification.

The thermal-vacuum environment will be applied to each of the 12 switches at one time. Subsequent to the thermal-vacuum test the switches will be divided into groups of two and tested as follows:

Switches	Environments
2 units	1, 2, 3, 5, 6
2 units	1, 2, 3, 6, 5
2 units	2, 3, 1, 5, 6

Switches	Environments
2 units	2, 3, 1, 6, 5
2 units	3, 1, 2, 5, 6
2 units	3, 1, 2, 6, 5

After the application of environments 1, 2, and 3, each switch will be operated for 100 cycles. Each switch will be operated for 1000 cycles under environment 5 and 1000 cycles under environment 6.

Batteries

Introduction

It is the purpose of this test plan to evaluate and demonstrate battery cell characteristics under a simulated Syncom II mission profile (launch and orbital operation).

The energy storage system for the Syncom II spacecraft includes four parallel nickel-cadmium batteries consisting of 24 hermetically sealed, series-connected cells. These batteries provide: 1) storage of electrical energy for operation during the launch phase and during eclipses, and 2) energy capacity for pulse loads such as the apogee motor igniter and the bipropellant reaction control system solenoids. These batteries are critical since they require: 1) a detailed specification and control of the active elements and mechanical properties such as the seal, and 2) a comprehensive selection and test program to meet the long lifetime requirement.

A minimum of 96 cells (equivalent of four Syncom II batteries) will be procured for the test program.

Applicable Documents

The following documents form a part of this test plan:

1) Battery Cell Procurement Specification.

2) Syncom II Environmental Requirements for Unit Qualification and Acceptance Tests.

3) Battery Cell Qualification and Acceptance Test Plan.

General Requirements

1) The test cells selected shall be characteristic and identical in configuration to all flight production units and shall be assembled, inspected, tested, and handled in the same manner as all flight production units.

2) The performance tests conducted prior to, during, and after environmental testing shall be in accordance with those specified during qualification and acceptance. The performance tests conducted as an integral part of this test phase shall include, but not be limited to:

   a) Effect of varying discharge currents on cell capacity.

   b) Effect of cycling at various depths and rates of end-of-charge and end-of-discharge voltages and cell capacity.

   c) Changing efficiency versus charge rate.

   d) False discharge.

   e) Thermal-vacuum overcharge-discharge.

Test Program

Test Samples. A representative sample of 64 two electrode cells and 32 three electrode (additional overcharge sensing electrode) cells will be selected for the test program. Acceptance tests will be initially performed on all test specimens in accordance with the acceptance test specifications. Subsequently, these will be divided into two groups of 48 assemblies each for qualification test cycling and accelerated life tests. The purpose of the qualification test cycling will be to demonstrate and ensure an adequate margin in design adequacy. Life tests will be performed to ensure compatibility of the cell design to the long lifetime requirement since these cells exhibit a wear-out or end-of-life characteristic rather than obeying a well defined exponential failure distribution.

Acceptance Tests. The 96 battery cells shall be divided into four groups of 24 cells each (16 two electrode cells and eight three electrode cells) and subjected to one cycle of each of the following acceptance level environments.

1) Shock and vibration

2) Thermal-vacuum and spin

3) Boost acceleration

4) Acoustical noise

Parameters of the test cycle environments shall be as specified by the Syncom II unit acceptance test plan and appropriate performance tests conducted prior to, during, and after exposure.

The four groups of cells shall be tested in the following order relative to environments 1, 2, 3, and 4.

Group I	1, 2, 3, 4
Group II	2, 3, 4, 1
Group III	3, 4, 1, 2
Group IV	4, 1, 2, 3

Upon completion of acceptance testing, one-half of the units from each group (8 two electrode and 4 three electrode cells) will be placed on an extended life test and the second half of each group will undergo qualification test cycling.

Qualification Test Cycling. Thirty-two two electrode cells and 16 three electrode cells selected upon completion of acceptance testing shall be subjected to one cycle of the following qualification level environments:

1) Shock and vibration

2) Thermal vacuum and spin

3) Boost acceleration

4) Acoustical noise

Parameters of the test cycle environments shall be as specified by the Syncom II unit qualification test plan and appropriate performance tests conducted prior to, during, and after exposure.

The battery cells selected for qualification test cycling shall be divided into four groups and tested in the following environmental sequence:

Group I	1, 2, 3, 4
Group II	2, 3, 4, 1
Group III	3, 4, 1, 2
Group IV	4, 1, 2, 3

Subsequent to group qualification testing, battery cells shall be selected from each group and subjected to additional cycles of each environment. For example, 8 two element and 4 three element cells may be selected from each group and exposed as follows:

12 cells - environment 1

12 cells - environment 2

12 cells - environment 3

12 cells - environment 4

Extended Life Test Program. Forty-eight Syncom II battery cells shall be subjected to concurrent temperature, vacuum-spin tests under the following conditions:

1) Test to be conducted for a period of 30 days.

2) During the 30-day period the temperature shall be cycled between the extreme high and extreme low values of the referenced acceptance test specifications. The temperature cycling shall include at least 200 cycles while simulating the expected temperature-time environment as far as practicable.

3) The spin rate and vacuum environments shall be in accordance with the referenced acceptance test specification.

The battery cell performance shall be monitored during these life tests as far as practicable. A complete performance check shall be made prior to and after all environmental exposures.

Following the environmental life tests, the same cells shall be operated with simulated spacecraft loading under room ambient conditions for a period of at least 1 year. Operational load cycling shall be programmed in accordance with expected values. Complete functional tests shall be conducted at specified intervals.

Sun Sensors

Scope

This test plan establishes the requirements and procedure for evaluation and demonstration of performance of the Syncom II sun sensor. Each sun sensor consists of a mechanical housing containing four solar cells which generate ψ pulses essential to spin rate control, orientation control, and antenna phase control. A minimum of eight solar sensors will be procured for the test program. The probability of success or reliability goal for each solar sensor has been established for each sun sensor as 0.998 for 1 year with reasonable confidence.

Applicable Documents

The following documents form a part of this test plan:

1) Hughes drawings: 496617-100 Sun Sensor Assembly, Multiple

2) Test Specifications (available at a later date): Environmental Requirements for Unit Qualification and Unit Acceptance Tests; and Sun Sensor Qualification and Acceptance Test Plan.

General Requirements

The specimens subjected to test shall be characteristic and identical in configuration to all flight production units and shall be assembled, inspected, tested, and handled in the same manner as all flight production units.

During shock-vibration and boost acceleration, testing each unit shall be energized with all rated power inputs applied.

During temperature-vacuum-spin testing, each unit shall be energized with all rated power inputs applied and with a light source of known intensity in an attempt to simulate sun input. Signal outputs shall be monitored for intermittencies or other indications of malfunctions.

Test Programs

Test Samples. A representative sample of eight solar sensor assemblies will be selected for the test program. Acceptance tests will be initially performed on the eight assemblies. Subsequently, these will be divided into two groups of four assemblies each for qualifications test cycling and accelerated life tests.

Acceptance Tests. The eight solar sensor assemblies shall be subjected to one cycle of each of the following acceptance level environments:

1) Shock and vibration

2) Thermal-vacuum and spin

3) Boost acceleration

4) Acoustical noise

Parameters of the test cycle environments shall be as specified by the Syncom II acceptance test plan referenced above.

A complete functional test shall be performed following each of the environmental test exposures tested in accordance with the acceptance test specification referenced above.

The eight sun sensor assemblies shall be tested in the following order relative to environments 1, 2, 3, 4.

Group I	1, 2, 3, 4
Group II	2, 3, 4, 1
Group III	3, 4, 1, 2
Group IV	4, 1, 2, 3

Upon completion of acceptance testing of all eight units, one unit shall be selected from each group to be placed on accelerated life testing, with the second unit from each group to undergo qualification test cycling.

Qualification Test Cycling. Four sun sensor assemblies shall be subjected to one cycle of each of the following qualification levels:

1) Shock and vibration

2) Temperature - vacuum-spin

3) Boost acceleration

4) Acoustical noise

Parameters of the test cycle environments shall be as specified by the qualification test plan previously referenced.

A complete functional test shall be performed, following each of the environmental test exposures in accordance with the qualification test specifications.

The four sun sensor assemblies shall be tested in the following order relative to environments 1, 2, 3, and 4.

1) First unit - 1, 2, 3, and 4

2) Second unit - 2, 3, 4, and 1

3) Third unit - 3, 4, 1, and 2

4) Fourth unit - 4, 1, 2, and 3

The four sun sensor assemblies shall then be subjected to additional cycles of the environments as follows:

1) The first unit to environment 1

2) The second unit to environment 2

3) The third unit to environment 3

4) The fourth unit to environment 4

Accelerated Life Test Program. Four sun sensor assemblies shall be subjected to concurrent temperature-vacuum-spin under the following conditions:

1) Test to be conducted over a period of 14 days.

2) During the 14-day period the temperature shall be cycled between the extreme high and extreme low values of the acceptance test specification. The total number of temperature cycles shall be 200.

3) The spin rate and vacuum shall be in accordance with the acceptance test specification.

Following the acceptance test program, the same four units shall be operated under laboratory ambient conditions for a period of at least 1 year. The operation is tentatively scheduled as follows: a simulated light source shall excite each sensor. The source shall be cycled on and off at a rate of 100 cycles per minute (simulating the spin rate of the spacecraft). The sun sensor performance will be monitored at selected intervals. (Four sources must operate for 1 year.)

Central Timers

Introduction

It is the purpose of this test plan to evaluate and demonstrate the central timer characteristics under a simulated Syncom II mission profile (launch and orbit operation).

The central timers (four per spacecraft) provide selectable outputs for the redundant phased array control electronics units at 2.81 minutes per pulse in addition to providing a means of firing the apogee engine.

The purpose of the tests is threefold: 1) to demonstrate the capability of the design to withstand stresses expected in the launch and space environments, 2) to demonstrate that there is no long time deterioration with life that would otherwise not be discovered before launch, and 3) to obtain data to determine what stress levels of certain types can be applied to timers scheduled for flight without damage to allow the culling of defective units.

The testing will be done in two phases:

Phase 1	Design tests - The testing of critical component parts of the timer.
Phase 2	Qualification tests - The testing of assembled timers.

During each phase, each part will be electrically energized to simulate actual conditions existing before and during flight and all electrical parameters will be monitored.

Applicable Documents

The following documents form a part of this test plan:

1) Central timer performance specification

2) Central timer qualification and acceptance test plans

Phase 1 - Design Tests

Two parts within the timer are considered to be critical components: 1) the incremental saturating cores (tape wound), and 2) the tuning fork oscillator. Samples (40 cores and 10 oscillators) of these parts will be subjected to the following stresses.

Encapsulation. This test will apply only to the incremental saturating cores since they are the only parts that must be encapsulated before being installed in the timer. If necessary, several encapsulating processes will be tested to determine the better process or processes for this type of circuitry. The better process or processes will be determined from the results of the stresses which follow.

Temperature Cycling (Nonoperating). All parts will be temperature cycled 100 times from -35 to + 85° C. The purpose is to cause poorly constructed parts to fail so that they may be culled.

Vibration and Shock. Sinusoidal and random vibration and shock stresses will be applied simultaneously. Step stresses of 1, 1-1/2, and 2 times the expected maximum stress will be applied. Thirty cores and eight oscillators willbe subjected to this test. Destruction tests will be attempted on one core from each group of encapsulation processes and one oscillator.

Acceleration. A centrifuge will simulate both boost and spin accelerations in a minimum of three axes. Step stresses of 1, 1-1/2, and 2 times the expected maximum stress will be applied. Twenty-nine cores and seven oscillators will be subjected to this test. Destructive tests will be attempted on one core from each group of encapsulating processes and one oscillator.

Thermal-Vacuum and Temperature Cycling. Approximately thirty cores and eight oscillators will be operated in a thermal-vacuum of 1×10^{-5} Torr for 30 days. The heat sink temperature will be cycled 200 times between the minimum and maximum temperature limits expected in orbit.

Operating Life Test. Ten cores and two oscillators previously segregated plus the cores and oscillators which survived the above stresses will be put on operating life test. This test will consist of electrically energizing each part to simulate actual operating conditions and monitoring the electrical parameters. The test duration will be a minimum of 3 months. Some parts will be in thermal-vacuum, some will be at atmospheric pressure. The test time in thermal-vacuum will be determined by the availability of the vacuum chamber.

Phase 2 - Qualification Tests

A sample lot of five timers will be used for these tests. Two timers will be segregated after temperature cycling (nonoperating) for operating life tests; the remaining three timers will be subjected to the tests listed below. One of the three timers may be used for destructive tests where applicable.

Temperature Cycling (Nonoperating). All timers will be temperature cycled 100 times from -35 to +85° C. The purpose is to cause poorly constructed timers to fail so that they may be culled.

Radio Frequency Noise. Three timers will be subjected to radio frequency noise simulating the worst conditions expected during the launch phase.

Vibration. Sinusoidal and random vibration and shock stresses will be applied simultaneously. Step stresses of 1, 1-1/4, and 2 times the expected maximum stress will be applied to the three timers.

Acceleration. A centrifuge will simulate both boost and spin accelerations in a minimum of three axes. Step stresses of 1, 1-1/4, and 2 times the expected maximum stress will be applied to the three timers.

Acoustical Noise. The three timers will be subjected to noise levels not to exceed 150 db.

Nuclear Radiation. During flight, two sources of radiation will be encountered: 1) Van Allen belt radiation, and 2) solar flare radiation. Van Allen belt radiation includes both electron and proton bombardment, while solar flare radiation includes only proton bombardment. Due to the expense and lead-time involved in conducting the radiation tests, studies must be made to determine the feasibility of conducting such tests. Radiation tests will be conducted on the three timers if the conclusions of the studies indicate their merit.

Electrical Noise. The three timers will be tested to assure that electrical noise on power leads will not cause timer intermittent failure.

Start Tests. The three timers will be subjected to start tests to ensure that if power is removed and then power is reapplied very slowly, the oscillators will have sufficient loop gain to restart oscillation.

Thermal-Vacuum and Temperature Cycling. The three timers will be operated in a thermal-vacuum of 1×10^{-5} Torr for 30 days. The heat sink temperature will be cycled 200 times between the minimum and maximum temperature limits expected in orbit.

Operating Life Test. The two timers previously segregated plus the timers which survived the above stresses will be put on operating life test. This test will consist of electrically energizing each timer to simulate actual operating conditions and monitoring the electrical parameters. The test duration will be a minimum of 3 months. Some timers will be in thermal-vacuum; some will be at atmospheric pressure. The test time in thermal-vacuum will be determined by the availability of the vacuum chamber.

Traveling-Wave Tubes

Introduction

It is the purpose of this test plan to evaluate and demonstrate the traveling-wave tubes (TWT) under simulated Syncom II mission profiles (launch and orbit operation).

There will be eight TWTs per spacecraft. A maximum of four will be operating at any given time, i.e., they are completely redundant.

Applicable Documents

The following documents form a part of this test plan:

1) TWT Performance Specification

2) TWT qualification and acceptance test plans.

General Requirements

The TWTs selected for this test shall be characteristic and identical to the flight production units and shall be assembled, inspected, tested, and handled in the same manner as all flight production units. There will be a total of 120 flight tubes built. Of these 120 tubes, 40 will be scheduled for flight use and 30 will be scheduled for the following test program.

Test Program

Twenty-four tubes will consist of a lot. After these tubes have passed the electrical performance requirements they will be subjected to a 2000-hour power aging test at laboratory ambient conditions. Tube parameters will be monitored at selected intervals during this 2000-hour test. At the conclusion of this test, eight flight tubes (complement for one spacecraft) will be selected based on the most stable operation during the 2000-hour test.

Five of the remaining 16 TWTs will be selected at random (the tubes must meet performance requirements) and be used for the following test. This group of five TWTs will be called group A. This procedure will be followed for the remaining four lots of 24 TWTs, except that 10 tubes will be selected from the final lot. The five tubes randomly selected from each succeeding lot will be labeled groups B, C, D. The ten tubes selected from the final lot will be labeled Group E.

The test program for the 30 traveling-wave tubes will be as follows:

Each of the 30 TWTs will be subjected to the expected launch environment which will consist of vibration (combined sinusoidal, random, and shock), acceleration (boost and spin), and temperature. Each of these tubes will then be subjected to 200 on-off cycles (both high voltage and filaments).

Group A TWTs will then be matched with flight-type power supplies and be subjected to 200 operational cycles of hot and cold temperatures (the minimum and maximum temperatures expected in orbit) under vacuum conditions (10^{-5} Torr or less). At the completion of the thermal-vacuum testing they will be operated (with flight power supplies) at laboratory ambient conditions for a minimum of 1 year. Tube and power supply performance will be monitored at selected intervals.

Group B TWTs will be operated at the minimum expected orbital temperature at laboratory ambient pressure for a minimum of 1 year. Tube performance will be monitored at selected intervals.

Group C TWTs will be operated at the maximum expected orbital temperature at laboratory ambient pressure for a minimum of 1 year. Tube performance will be monitored at selected intervals.

Group D and E TWTs will be operated at laboratory ambient conditions for a minimum of 1 year. Tube performance will be monitored at selected intervals.

For additional assurance it is planned to use electronically unacceptable tubes (those that would not meet the performance requirement prior to the 2000-hour power aging test) to determine the effects of acceleration (a) shock (s) and vibration (v). The experiment will involve two levels of each environment. The first level will be the qualification level and duration and the second level will be two times the qualification level and duration. The tests will be conducted according to the following matrix.

		S_1	S_2
a_1	V_1	2	2
	V_2	2	2
a_2	V_1	2	2
	V_2	2	2

After the tubes have been subjected to the environments in the above random manner they will be bench life tested.

The results will indicate the effects of a specific environment on tube life including any interactions.

ADVANCED BILL OF MATERIALS - Syncom II

The attached advanced bill of materials is for electronic components of the Syncom II spacecraft. The preferred parts list (located in Appendix B of this report) may be used to obtain the commercially equivalent part numbers that are associated with the Hughes part numbers.

As indicated on the advanced bill of materials, there are instances where the exact value of resistors and capacitors are not known. It is believed, however, that the total number of the spacific items, e.g., carbon composition resistors, deposited carbon, etc., is approximately 90 percent accurate if it were to be compared with the final bill of materials.

SEMICONDUCTORS Syncom II 30 April 1963	Units Per S/C	1N21F	914-1 1N198	704-4 1N748A	1N748	704-5 1N749A	704-12 1N756A	1N1736A	1N2810	718-1 1N3062	714-1 1N3189	817-1 1N3257	2N491	2N683	819-1 2N707A	801-1 2N722	2N684	829-1 2N1141	828-1 2N1405	802-1 2N871	826-1 2N1506	813-1 2N1709	816-1 2N1724	838 2N1890	824-2 2N2183A	847 2N2303
Master oscillator 66 mc SP190013	4			4											4				8		4					
Tripler SP190004	8																									
Transmitter master oscillator SP190011	4																	4	4							
X32 multiplier SP190023	12																									
X2 multiplier SP190024	8																									
Inverter SP168564	4																			8						
Phase modulator SP190009	4			4								16						8	8							
IF post amplifier SP190007	4					4													24							
TWT power supply SP190021	8															8							16			
IF wide band limiter SP190018	4						4											12								
IF preamplifier 54 mc SP172029	4					4													12							
Multiplier amplifier SP190010	4			4											4			8	4							
Input mixer SP167565	4																									
High level mixer SP172382	4	4																								
Beacon oscillator SP190012	4			4															12							
IF amplifier SP190022	4																		12							
Receiver master oscillator 66.232mc SP190014	4			4											4				8							
Command receiver 475210	4		4	4						24						4			36							
Telemetry transmitter 475220	4			8											4				12		4	4				
Transponder regulator 475101	4															16										
Telemetry regulator 475825	4															16										
Transmitter regulator 475175	8															16										
Diplexer 475201	4																									
Command regulator 475212	4															8										
Solar panel 475252	14										112															
Solar panel, special 475253	2										16												8			
Battery regulator 475251	4							8	8				4				4									
Wiring harness	1																									
Pace (see Note 2)	4																			116				64		132
Timer and apogee driver														70											80	18
Command encoder																				50						18
Jet control electronic - solenoid											16									16						32
drive																										
Telemetry decoder																										
Regulator phased array (dig.) 475160	4																									
Regulator phased array (anal) 475163	1																									
Number per spacecraft		4	4	32		8	4	8	8	24	144	16	4	70	16	52	4	32	140	190	8	4	24	64	80	200

SEMICONDUCTORS Syncom II 30 April 1963	2N2303A	2N2484	2N2489	2N2342	705-1	705-2	713-1 FD300	Z1902-1	Z2	918-2 ma4325F	918-1 ma4325D	ma4330A	905-6 ma4355A	905-B ma4355C	903-3 PC4002	903-7 PC4005	903-9 PC4006	724-1 RD500	HD18110	D4142M	UT118	SP8414A	18-20V ZENER	2N2185	801-1 2N722
Master oscillator 66 mc SP190013												16		16											
Tripler SP190004																									
Transmitter master oscillator SP190011										24	12		24	24			24								
X32 multiplier SP190023												16													
X2 multiplier SP190024																									
Inverter SP168564																		16							
Phase modulator SP190009																									
IF post amplifier SP190007																									
TWT power supply SP190021																			8						
IF wide band limiter SP190018																									
IF preamplifier 54 mc SP172029																									
Multiplier amplifier SP190010																									
Input mixer SP167565																									
High level mixer SP172382																									
Beacon oscillator SP190012																									
IF amplifier SP190022																									
Receiver master oscillator 66.232mc SP190014																									
Command receiver 475210															16	8		4							
Telemetry transmitter 475220																									
Transponder regulator 475101																									
Telemetry regulator 475825																									
Transmitter regulator 475175																									
Diplexer 475201																									
Command regulator 475212																									
Solar panel 475252																									
Solar panel, special 475253				8																	8				
Battery regulator 475251																									
Wiring harness																									
Pace (see Note 2)	380	544			4	4	3480	236	4													80			
Timer and apogee driver	22	19					68	14															70		
Command encoder	200	335					2100	60																	
Jet control electronic - solenoid		60	24				452																		
drive	72	40			4		1692	16	4															340	
Telemetry decoder																									8
Regulator phased array (dig.) 475160																									2
Regulator phased array (anal) 475163																									
Number per spacecraft	674	998	24	8	8	4	1792	336	8	24	12	32	24	40	16	8	24	20	8		8	80	70	340	10

Note: 1. The Hughes part number (988XXX) is listed whenever possible. 988 is omitted, only the last three numbers and the dash number are used.

2. Detailed list on Page 3.

SEMICONDUCTORS Syncom II 30 April 1963		Units per S/C	844-1	704-10	700-1	719-1	719-2	817-2	704-14	712-	710-1	702-1	703-1	802-3	723-1	712-7
Master oscillator	SP190013	4														
Tripler	SP190004	8														
Transmitter master oscillator	SP190011	4														
X32 multiplier	SP190023	12														
X2 multiplier	SP190024	8														
Inverter	SP168564	4														
Phase modulator	SP190009	4														
IF postamplifier	SP190007	4														
TWT power supply	SP190021	8	16	8	16	32	64									
IF wideband limiter	SP190018	4														
IF preamplifier 54 mc	SP172029	4														
Multiplier amplifier	SP190010	4														
Input mixer	SP167565	4														
High level mixer	SP172382	4														
Beacon oscillator	SP190012	4														
IF amplifier	SP190022	4														
Receiver master oscillator 66.232mc	SP190014	4														
Command receiver	475210	4														
Telemetry transmitter	475220	4														
Transponder regulator	475101	4		8				8	8							
Telemetry regulator	475822	4		8				8	8							
Transmitter regulator	475175	8		8				8	8	8	8					
Diplexer	475201	4														
Command regulator	475212	4	8	4	4			4	4			4	8			
Solar panel	475252	14														
Solar panel, special	475253	2														
Battery regulator	475251	4														
Wiring harness		1														
Pace (see Note 2)		4														
Timer and apogee driver		1														
Command decoder		1														
Jet control electronic - solenoid drive		1														
Telemetry encoder		1														
Regulator phased array (dig)	475160	4	8	1	1			4	1			1	8			
Regulator phased array (anal)	475163	1		1	1			3	1		1		2	1	4	1
Number per spacecraft			32	38	22	32	64	35	30	8	9	5	18	1	4	1

Note: 1. The Hughes part number (988XXX) is listed whenever possible. 988 is omitted, only the last three numbers and the dash number are used.

2. Detailed list on Page 3

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SEMICONDUCTORS Syncom II 30 April 1963	Units per S/C	FD-100 988713-1	2N2484	2N871 988802	2N2303A	SF8414A	Z1	2N2303	2N1890	988705-1	988705-2	988817-2	988801-1	988844-1	988704-10	988704-14	988702-1	988700-1	988703-1	988802-3	98872C-1	988712-7	988710-1	Z2
PACE																								
ψ and ψ_2 amplifier	2	8	8	6			1																	
Standard flip-flops	36	72	72																					
Voltage-controlled oscillator	1	3	4		2		3			1	1													
Error pulse gate	1		2		2																			
Lock F-F logic (F702)	1	29	4																					
FF - F110	1	17	4																					
FF - F100	1	5	4																					
Reset buffer (F301)	1	10	1																					
J100, J101	2	16	4																					
FFA111	1	2	2																					
	1																							
Reset amplifier #2 (B301)	1	13		1																				
Reset amplifier #3 (B300)		10		1																				
Transfer gate		9																						
Standard inverter	6	24	6																					
BVW switch (540X, 541X)	16	128	16					16																
BVW ladder	2																							
Interstage gate	32	32																						
Diode function generator	2	24																						
Inverting amplifier	2	2		2	4	2	2																	
DC amplifier	2	2	2	2	6	2	2																	
Vector summing network	4			16	16																			
Generator I (sin)	8	120					24																	
Generator II (cos)	8	96					8																	
General auxiliary supplies	1																							
Power amplifiers	16	16			16	16	16	16	16															
Timer selection switch	1	12	1																					
Digital readout encoder	4	192			48																			
-2.3 volt supply	1	1		1	1		1																	1
+12 volt supply	1	1	1		1																			
+6 volt supply	1	1	1		1		1																	
-5 volt supply	1	1					1	1																
Inverting Schmidt	2	24	4																					
B101 gate	1	2																						
Number per pace		870	136	29	97	20	59	33	16	1	1													1

RESISTORS Syncom II 30 April 1963		Units per S/C	601-74	601-81	601-126	602-66	602-69	623-161	623-166	623-175	623-183	623-201	623-205	623-251	623-321	623-325	623-351	640-296	640-442	640-532	640-364	640-544	601-	602-	604-	610-	623-
Master oscillator, 66 mc	SP190013	4																									
Tripler	SP190004	8																									
Transmitter master oscillator	SP190001	4																									
X32 multiplier	SP190023	12																									
X2 multiplier	SP190024	8																									
Inverter	SP168564	4																									
Phase modulator	SP190009	4																									
IF postamplifier	SP190007	4																									
TWT power supply	SP190021	8																									
IF wide band limiter	SP190018	4																									
IF preamplifier, 54 mc	SP172029	4																									
Multiplier amplifier	SP190010	4																									
Input mixer	SP167565	4																									
High level mixer	SP172382	4																									
Beacon oscillator	SP190012	4																									
IF amplifier	SP190022	4																									
Receiver master oscillator, 66.232mc	SP190014	4																									
Command receiver	475210	4																									
Telemetry transmitter	475220	4																									
Transponder regulator	475101	4																									
Telemetry regulator	475222	4																									
Transmitter regulator	475175	8																									
Diplexer	475201	4																									
Command regulator	475212	4																									
Solar panel	475252	14																									
Solar panel, special	475253	2																									
Battery regulator	475253	4																									
Wiring harness		1																									
Pace (see Note 2)		4	4	16	28	4	4	8	4	4	4	4	8	9	8	8	4		16	8	4	4					216
Timer and apogee driver																											
Command decoder																											
Jet control electronic - solenoid drive																											
Telemetry encoder																							36	8		16	
Regulator phased array (dig.)	475160	4																					12		2	4	
Regulator phased array (anal)	475163																										
Number per spacecraft			4	16	28	4	4	8	4	4	4	4	8	9	8	8	4		16	8	4	4		8	2	20	216

Note: 2. Detailed list on Page 3.

RESISTORS
Syncom II 30 April 1963

		Units per S/C	601-101	601-102	601-103	601-104	601-105	601-107	601-108	601-109	601-110	601-111	601-113	601-115	601-116	601-118	601-119	601-121	601-124	601-125	601-129	601-132	601-133	601-134	601-135	601-137	601-147	601-128	601-148
Master oscillator, 66 mc	SP190013	4																											
Tripler	SP190004	8																											
Transmitter master oscillator	SP190011	4																											
X32 multiplier	SP190023	12																											
X2 multiplier	SP190024	8																											
Inverter	SP168564	4						4																					
Phase modulator	SP190003	4																											
IF postamplifier	SP190007	4																											
TWT power supply	SP190027	8																											
IF wide band limiter	SP190018	4																											
IF preamplifier, 54 mc	SP172029	4																											
Multiplier amplifier	SP190010	4																											
Input mixer	SP167565	4																											
High level mixer	SP172382	4																											
Beacon oscillator	SP190012	4																											
IF amplifier	SP190022	4																											
RCVR master oscillator, 66.223mc	SP190014	4																											
Command receiver	475210	4										4																	
Telemetry transmitter	475220	4																	4									4	
Transponder regulator	475222	4																											
Telemetry regulator	475222	4																											
Transmitter regulator	475175	8																											
Diplexer	475201	4																											
Command regulator	475212	4																											
Solar panel	475252	14																											
Solar panel, special	475253	2																											
Battery regulator	475251	4																											
Wiring harness		1																											
Pace (see page 4 for detail list)		4	12	40	12	76	132		8	416	336	92	4	208	8	348	12	4	76	64	8	4	4	8	12	8	8	8	4
Timer and apogee driver																													
Command decoder																													
Jet control electronic - solenoid drive																													
Telemetry encoder																													
Total per spacecraft			12	40	12	76	132	4	8	416	336	96	4	208	8	348	12	4	76	64	8	4	4	8	12	8	8	8	4

		Units per S/C	602-32	602-46	602-53	602-55	602-56	602-68	602-79	602-87	602-88	603-15	603-37	603-56	603-77	610-235	610-262		611-213	611-300		601-53
Master oscillator, 66 mc	SP190013	4																				
Tripler	SP190004	8																				
Transmitter master oscillator	SP190011	4																				
X32 multiplier	SP190023	12																				
X2 multiplier	SP190024	8																				
Inverter	SP168564	4																				
Phase modulator	SP190003	4			8		8				4				4							
IF postamplifier	SP190007	4																				
TWT power supply	SP190027	8																				
IF wide band limiter	SP190018	4			8		12	4						4							4	
IF preamplifier, 54 mc	SP172029	4																				
Multiplier amplifier	SP190010	4			8		12		4						4							
Input mixer	SP167565	4																				
High level mixer	SP172382	4																				
Beacon oscillator	SP190012	4																				
IF amplifier	SP190022	4																				
RCVR master oscillator, 66.223mc	SP190014	4																				
Command receiver	475210	4													4							
Telemetry transmitter	475220	4	8	8								8	4	4	4							
Transponder regulator	475222	4																				
Telemetry regulator	475222	4																				
Transmitter regulator	475175	8																				
Diplexer	475201	4																				
Command regulator	475212	4																				
Solar panel	475252	14																	14	14		
Solar panel, special	475253	2																	2	2		
Battery regulator	475251	4				4				4												
Wiring harness		1																				
Pace (see page 4 for detail list)		4														8	8					
Timer and apogee driver																						
Command decoder																						
Jet control electronic - solenoid drive																						
Telemetry encoder																						
Total per spacecraft			8	8	24	4	32	4	4	4	4	8	4	8	16	8	8		16	16	4	

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		Units per S/C	601-7	601-15	601-17	601-23	601-25	601-27	601-28	601-29	601-31	601-32	601-33	601-34	601-39	601-43	601-46	601-51
Master oscillator, 66mc	SP190013	4										8						
Tripler	SP190004	8																
Transmitter oscillator	SP190011	4					8	4				8						
X32 multiplier	SP190023	12																
X2 multiplier	SP190024	8																
Inverter	SP168564	4																
Phase modulator	SP190009	4												8				
IF postamplifier	SP190007	4										20						
TWT power supply	SP190021	8																
IF wide band limiter	SP190018	4				4							4		8			4
IF preamplifier, 54 mc	SP172029	4								4		8						
Multiplier amplifier	SP190010	4		8							8							
Input mixer	SP167565	4																
High level mixer	SP172382	4																
Beacon oscillator	SP190012	4													20			
IF amplifier	SP190022	4																
Receiver master oscillator	SP190014	4										8						
Command receiver	475210	4			4	4	8	4								20	8	
Telemetry transmitter	475220	4		8								8	8			8		
Transponder regulator	475101	4																
Telemetry regulator	475222	4																
Transmitter regulator	475175	8																
Diplexer	475201	4																
Command regulator	475212	4																
Solar panel	475252	14																
Solar panel, special	475253	2																
Battery regulator	475251	4																
Wiring harness		1																
Pace (see page 5 for detail list)		4													8		8	
Timer and apogee driver																		
Command decoder																		
Jet control electronic - solenoid drive																		
Telemetry encoder																		
Regulator phased array (dig.)	475160	4																
Regulator phased array (anal.)	475163	1																
Total per spacecraft				16	4	8	16	8		4	8	60	12	8	36	28	16	4

	601-55	601-56	601-57	601-58	601-60	601-61	601-63	601-65	601-66	601-67	601-69	601-70	601-71	601-73	601-74	601-75	601-77
Master oscillator, 66mc							4					8					8
Tripler																	
Transmitter oscillator					4		4	4									4
X32 multiplier																	
X2 multiplier																	
Inverter		8															
Phase modulator							4			32		16					
IF postamplifier							20					24					
TWT power supply																	
IF wide band limiter	8					4						8				4	
IF preamplifier, 54 mc							12					12					
Multiplier amplifier												4	12			4	
Input mixer																	
High level mixer																	
Beacon oscillator							4					12					8
IF amplifier		4													8	4	
Receiver master oscillator		4					4					8					8
Command receiver			4				4			4			24	8	4	24	
Telemetry transmitter				4			12				12			4			
Transponder regulator																	
Telemetry regulator																	
Transmitter regulator																	
Diplexer																	
Command regulator																	
Solar panel																	
Solar panel, special																	
Battery regulator																	
Wiring harness																	
Pace (see page 5 for detail list)	4	28				8			8	40		4	72	8			8
Timer and apogee driver																	
Command decoder																	
Jet control electronic - solenoid drive																	
Telemetry encoder																	
Regulator phased array (dig.)																	
Regulator phased array (anal.)																	
Total per spacecraft	12	44	4	4	4	12	68	4	8	76	12	96	108	20	12	36	32

	601-79	601-80	601-83	601-85	601-86	601-87	601-88	601-89	601-90	601-91	601-92	601-93	601-94	601-95	601-97	601-98	601-99	601-100
Master oscillator, 66mc				4							4							
Tripler																		
Transmitter oscillator																		
X32 multiplier																		
X2 multiplier																		
Inverter											8							
Phase modulator	20																	
IF postamplifier		4																
TWT power supply																		
IF wide band limiter											4							
IF preamplifier, 54 mc		4																
Multiplier amplifier																		
Input mixer																		
High level mixer																		
Beacon oscillator	4																	
IF amplifier	8					4												
Receiver master oscillator																		
Command receiver		4	8		4	32		4										
Telemetry transmitter						24												
Transponder regulator																		
Telemetry regulator																		
Transmitter regulator																		
Diplexer																		
Command regulator																		
Solar panel																		
Solar panel, special																		
Battery regulator																		
Wiring harness																		
Pace (see page 5 for detail list)	8			72		148	4	40	4	8		96	64	4	68	4	40	8
Timer and apogee driver																		
Command decoder																		
Jet control electronic - solenoid drive																		
Telemetry encoder																		
Regulator phased array (dig.)																		
Regulator phased array (anal.)																		
Total per spacecraft	40	12	8	76	4	208	4	44	4	8	16	96	64	4	68	4	40	8

RESISTORS Syncom II 30 April 1963	Units per S/C	620-167	620-178	620-179	620-190	620-191	620-222	620-299	621-193	623-16	623-27	623-38	623-63	623-69	623-70	623-72	623-103	623-113	623-163	623-177	623-229	623-231	623-232	623-237	623-249	623-264	623-276	623-289	623-296	623-303	623-318	623-319	623-329	623-332	623-343	623-356	623-361	623-365	623-371	623-385	623-390	601- *	602- *	603- *	604- *	610- *	623- *
Master oscillator, 66 mc SP190013	4																																														
Tripler SP190004	8																																														
Transmitter master oscillator SP190017	4																																														
X32 multiplier SP190023	2																																														
X2 multiplier SP190021	8																																														
Phase modulator SP190009	4																																														
IF postamplifier SP190007	4																																									32		8	8		
TWT power supply SP190021*	8																																														
IF wide band limiter SP190018	4																																														
IF preamplifier, 54 mc SP172029	4																																														
Multiplier amplifier SP190010	4																																														
Input mixer SP167505	4																																														
High level mixer SP172382	4																																														
Beacon oscillator SP190012	4																																														
IF amplifier SP190022	4																																														
Receiver master oscillator SP190014	4																																														
Command receiver 475210	4																																														
Telemetry transmitter 475220*	4																																									56	8			24	
Transponder regulator 475101*	4																																									56	8			24	
Telemetry regulator 475222*	4																																									96			16	32	
Transmitter regulator 475175*	8																																														
Diplexer 475201*	4																																									36	8			16	
Command regulator 475212*	4																																														
Solar panel 475252	14																																														
Solar panel, special 475253	2																																														
Battery regulator 475251	4																																														
Wiring harness	1																																														
Pace (see Page 4 for detail list)	4	8	8	8	8	16	8	8	72	32	32	64	4	32	16	16	32	8	16	16	16	64	8	64	16	16	4	8	32	16	32	128	64	32	8	8	8	32	8	104	16	48	8		2	20	
Timer and apogee driver*																																										370					
Command decoder*																																										1600					90
Jet control electronic - solenoid drive*																																										412		16	16		
Telemetry encoder*																																										664					72
*Estimated																																															
Total per spacecraft		8	8	8	8	16	8	8	72	32	32	64	4	32	16	16	32	8	16	16	16	64	8	64	16	16	4	8	32	16	32	128	64	32	8	8	8	32	8	104	16	3370	32	24	42	116	162

RESISTORS Syncom II PACE 30 April 1963	Units per S/C	601-39	601-46	601-55	601-56	601-61	601-66	601-67	601-70	601-71	601-73	601-77	601-79	601-85	601-87	601-88	601-89	601-90	601-91	601-93	601-94	601-95	601-97	601-98	601-99	601-100	601-101	601-102	601-103	601-104	601-105	601-108	601-109	601-110	601-111	601-113	601-115	601-116	601-118	601-119	601-121	601-124	601-125	601-129	601-132	601-133	601-134	601-135	601-137	601-147	601-128	601-148
ψ_1 and ψ_2 amplifier	2	2				2					2		2								2				2	2	2		2					2						2												
Standard flip-flops	36																																72	72					72													
VCO	1			1					1					1	1							1													2			1		1								1	2		1	
Error pulse gate	1						2					1				1	1							1	1		1	2					1						1													
Lock F-F logic (F702)	1																				1				1							1	2			1	1		1							1	2				1	
F-F - F110	1																							2											4		2															
F-F - F100	1																							2											3		2															
Reset buffer (F301)	1																																		2				1						1							
J100, J101	2																								4								4																			
FFA111	1																																1				1		1		1											1
Reset amplifier #2 (A301)	1																				1													1			1															
Reset amplifier #3 (B300)	1																				1													1			1															
Transfer gate	1																				9																															
Standard inverter	6							6																													12															
BVW switch (540X, 541X)	16																			16			16							16							32					16										
BVW ladder	2																																																			
Interstage gate	32																														32																					
Diode function generator	2																																																			
Inverting amplifier	2				2			4							2																																					
Amplifier, DC	2				4										2				2						2																	2								2		
Vector summing network	4													16	16																		16																			
Generator I (sin)	8																			8													8	8	8				8				8									
Generator II (cos)	8																8											8															8									
General auxiliary supplies	1																																																			
Power amplifiers	16									16					16																																					
Timer selection switch	1																																		4													2				
Digital readout encoder	4																																																			
-2.3 volt regulator	1													1							1									1									1													
+12 volt regulator	1																				1		1						1									1														
+6 volt (a) regulator	1											1																														1										
+6 volt (b) regulator	1				1												1	1													1	1																				
Inverting Schmidt	2		2							2																				2														2								
B101 gate																																							2													
Total per page		2	2	1	7	2	2	10	1	18	2	2	2	18	37	1	10	1	2	24	16	1	17	5	10	2	3	10	3	19	33	2	104	84	23	1	52	2	87	3	1	19	16	2	1	1	2	3	2	2	2	1

RESISTORS Syncom II PACE 30 April 1963	Units per S/C	610-295	610-262	620-167	620-178	620-179	620-190	620-191	620-222	620-299	621-193	623-16	623-27	623-38	623-63	623-69	623-70	623-72	623-103	623-113	623-166	623-177	623-229	623-231	623-237	623-249	623-264	623-276	623-289	623-296	623-303	623-318	623-319	623-329	623-332	623-343	623-361	623-365	623-371	623-386	623-340	623-232	623-356	601-	602-	604-	610-	623-
ψ and ψ_2 amplifier	2	2	2					4	2											2																2			2			2						
Standard flip-flops	36																																															
Voltage-controlled oscillator	1									2																																						
Error pulse gate	1																																															
Lock F-F logic (F702)	1																																															
F-F F110	1																																															
F-F F100	1																																															
Reset buffer (F301)	1																																															
J100, J101	2																																															
F-F A111	1																																															
Reset amplifier #2 (A301)	1																																															
Reset amplifier #3 (B300)	1																																															
Transfer gate	1																																															
Standard inverter	6																																															
BVW switch (540X, 541X)	16													16																										14								
BVW ladder	2										18																																					
Interstage gate	32																																															
Diode function generator	2			2	2	2	2										4	4			4	4	4			4	4										2											
Inverting amplifier	2																																					4		4			2					
dc amplifier	2																												2									4										2
Vector summing network	4																													8	4	8			8					8	4							
Generator I (sin)	8											8							8																													16
Generator II (cos)	8												8			8																																24
General auxiliary supplies	1														1													1																				12
Power amplifiers	16																							16	16								32	16														
Timer selection switch	1																																															
Digital readout encoder	4																																															
-2.3 volt regulator	1																																															
+12 volt regulator	1																																															
+6 volt (a) regulator	1																															1															
+6 volt (b) regulator	1																																															
Inverting Schmidt	1																																															
B101 gate																																																
Total per pace		2	2	2	2	2	2	4	2	2	18	8	8	16	1	8	4	4	8	2	4	4	4	16	16	4	4	1	2	8	4	8	33	16	8	2	2	8	2	26	4	2	2					54

RESISTORS Syncom II PACE 30 April 1963	Units per S/C	601-74	601-81	601-126	602-66	602-69	623-161	623-165	623-175	623-183	623-201	623-205	623-251	623-321	623-325	623-351	640-296	640-442	640-537	640-364	640-544
ν_1 and ν_2 amplifier	2																				
Standard flip-flops	36																				
Voltage-controlled oscillator	1																				
Error pulse gate	1																				
Lock F-F logic (F702)	1																				
F-F F110	1																				
F-F F100	1																				
Reset buffer (F301)	1																				
J100, J101	2																				
FF A111	1																				
Reset amplifier #2 (A301)	1																				
Reset amplifier #3 (B300)	1																				
Transfer gate	1																				
Standard inverter	6			6																	
BVW switch (540X, 541X)	16																				
BVW ladder	2																				
Interstage gate	32																				
Diode function generator	2																				
Inverting amplifier	2																				
Amplifier, dc	2																				
Vector summing network	4																				
Generator I (sin)	8																				
Generator II (cos)	8																				
General auxiliary supplies	1						2	1	1	1	1	2		2	2	1					
Power amplifier	16																				
Timer selection switch	1																				
Digital readout encoder	4																				
-2.3-volt regulator	1		1			1												1		1	
+12-volt regulator	1			1	1								1					1			1
+6-volt (a) regulator	1	1	2															1	1		
+6-volt (b) regulator	1		1															1	1		
Inverting Schmidt	2																				
B101 gate																					
Total per pace		1	4	7	1	1	2	1	1	1	1	2	1	2	2	1		4	2	1	1

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CAPACITORS Syncom II 30 April 1963	Units per S/C	500-4	500-8	500-53	500-55	500-56	500-72	500-78	500-152	500-156	500-186	500-205	500-239	500-160	501-7	502-3	502-10	502-31	502-54
Master oscillator, 66 mc SP190013	4						4												
Tripler SP190004	8																		
Transmitter oscillator SP190011	4																		
X32 multiplier SP190023	12																		
X2 multiplier SP190024	8																		
Inverter SP166564	4	20	4																
Phase modulator SP190009	4												8						
IF postamplifier SP190007	4								4										
TWT power supply SP190021	8																		
IF wide band limiter SP190018	4											4							
IF preamplifier, 54 mc SP172029	4								4										
Multiplier amplifier, X2 SP190010	4												4						
Input mixer SP167565	4																		
High level mixer SP172382	4																		
Beacon oscillator SP190012	4						4												
IF amplifier SP190022	4																		
Receiver master oscillator SP190014	4													4					
Command receiver 475210	4				4		4												
Telemetry transmitter	4						4								4				
Transponder regulator	4																		
Telemetry regulator	4																		
Transmitter regulator	8																		
Diplexer	4																		
Command regulator	4																		
Solar panel	14																		
Solar panel, special	2																		
Battery regulator	4									8									
Wiring harness	1																		
Pace (see Page 3 for detailed list)	4	12		32		40		8			8					4	4	8	8
Timer and apogee driver	1																		
Command decoder	1																		
Jet control electronic - solenoid drive	1																		
Telemetry encoder	1																		
Regulator phased array (dig.) 475160	4																		
Regulator phased array (anal) 475163	1																		
Total per spacecraft		32	4	32	4	40	16	8	8	8	8	4	12	4	4	4	4	8	8

CAPACITORS Syncom II 30 April 1963	503-12	503-14	503-19	503-21	503-23	503-27	503-35	503-36	503-37	503-42	503-43	503-46	503-48	503-50	503-54	503-59	503-60	503-61	503-67	503-74	503-107	503-115	503-126	503-127	503-129	503-133	503-149	503-153
Master oscillator, 66 mc SP190013																						8						
Tripler SP190004																												
Transmitter oscillator SP190011											4																	
X32 multiplier SP190023																												
X2 multiplier SP190024																												
Inverter SP166564																			4									
Phase modulator SP190009												4	8															
IF postamplifier SP190007																												
TWT power supply SP190021																												
IF wide band limiter SP190018																						4						
IF preamplifier, 54 mc SP172029																												
Multiplier amplifier, X2 SP190010							4			4																		
Input mixer SP167565																												
High level mixer SP172382																												
Beacon oscillator SP190012			4	4																		4						
IF amplifier SP190022			8			4													4									
Receiver master oscillator SP190014											4																	
Command receiver 475210	8		12			4		4						24			4		8									
Telemetry transmitter	4	4	12		4	4	8		12		8					8			24									
Transponder regulator																												
Telemetry regulator																												
Transmitter regulator																												
Diplexer																												
Command regulator																												
Solar panel																												
Solar panel, special																												
Battery regulator																												
Wiring harness																												
Pace (see Page 3 for detailed list)			16												72		36	40		4	4		8	4	4	288	4	8
Timer and apogee driver																												
Command decoder																												
Jet control electronic - solenoid drive																												
Telemetry encoder																												
Regulator phased array (dig.) 475160																												
Regulator phased array (anal) 475163																												
Total per spacecraft	12	4	52	4	4	12	12	4	12	4	16	4	8	24	72	8	40	40	40	4	4	16	8	4	4	288	4	8

CAPACITORS Syncom II 30 April 1963		Units per S/C	504-12	504-16	504-28	504-32	505-7	505-24	505-58	520-1	520-2	526-1	541-30	704-5	CB11RE102J	2404-041-102	500-18	500-107	300-69	.05hf 1% now	*500-	*501-	*502-	*503-	*525-
Master oscillator, 66 mc	SP190013	4			36					16		28													
Tripler	SP190004	8								24					32	32									
Transmitter oscillator	SP190011	4			28					8		24													
X32 multiplier	SP190023	12								156					120	72									
X2 multiplier	SP190024	8																							
Inverter	SP168564	4		4																					
Phase modulator	SP190009	4	56							12	8	36													
IF postamplifier	SP190007	4	24									76													
TWT power supply	SP190021	8																			16	72		8	
IF wide band limiter	SP190010	4			44							24													
IF preamplifier, 54 mc	SP172028	4			8							40													
Multiplier amplifier, X2	SP190010	4			90					12															
Input mixer	SP167565	4																							
High level mixer	SP172382	4																							
Beacon oscillator	SP180032	4			36					12															
IF amplifier	SP190022	4								16		32													
Receiver master oscillator	SP190014	4			36					8		12	4												
Command receiver	475210	4				104				44		100													
Telemetry transmitter	475220	4				48				32	8	36													
Transponder regulator	475101	4																			8			16	
Telemetry regulator	475222	4																			8			16	
Transmitter regulator	475175	8																			8			16	
Diplexer	475201	4																							
Command regulator	475212	4																			12	8		12	
Solar panel	475252	14																							
Solar panel, special	475253	2																							
Battery regulator	475251	4					4	4	4																
Wiring harness		1																							
Pace		4															12	8		8					
Timer and apogee driver*		1																			32			150	
Command decoder*		1																			106			150	
Jet control electronic - solenoid drive*		1																			8			8	
Telemetry encoder*		1																			32		12	52	60
Regulator phased array (dig.)	475160*	4																			12	8		12	
Regulator phased array (anal)	475163*	1																			3	2		3	
Total per spacecraft			80	4	228	152	4	4	4	340	16	408	4		152	104	12	8		8	245	90	12	443	60

*Estimates

CAPACITORS Syncom II PACE 30 April 1963	Units per S/C	500-4	500-53	500-56	500-78	500-186	502-3	502-10	502-31	502-54	.05 uf 100V	503-19	503-54	503-60	503-61	503-74	503-107	503-126	503-127	503-129	503-133	503-149	503-153	500-48	500-107	500-69	500-	501-	502-	503-	525-
ψ and ψ2 amplifier	2		8	10	2	2					2																				
Standard flip-flops	36																				72										
Voltage-controlled oscillator	1	3					1	1		2																					
Error pulse gate	1																														
Lock F-F logic (F702)	1																1		1	1			2								
F-F F-110	1											2			2																
F-F F-100	1											2			2																
Reset buffer (F301)	1																					1									
J100, J101	2														4																
F-F A111	1															1															
Reset amplifier No. 2 (A301)	1	none																													
Reset amplifier No. 3 (B300)	1	none																													
Transfer gate	1													9																	
Standard inverter	6	none																													
BVW switch (540X, 541X)	16	none																													
BVW ladder	2	none																													
Interstage gate	32	none																													
Diode function generator	2								2																						
Inverting amplifier	2												2																		
Amplifier dc	2																	2													
Vector summing network	4	none																													
Generator I (sin)	8	none																													
Generator II (cos)	8	none																													
General auxiliary supplies	1	none																													
Power amplifiers	16												16																		
Timer selection switch	1	none																													
Digital readout encoder	4	none																													
-2.3-volt regulator	1																							1							
+12-volt regulator	1																								1						
+6-volt (a) regulator	1																							1	1						
+6-volt (b) regulator	1																							1							
Inverting Schmidt	2																														
B101 gate															2																
Number per pace		3	8	10	2	2	1	1	2	2	2	4	18	9	10	1	1	2	1	1	72	1	2	3	2		75	10		15	

COIL, CORES, CHOKES, AND TRANSFORMER Syncom II 30 April 1963		Units per S/C	Core T37-4	Core T37-6	Core T50-2	Core T50-3	Core T66-6	457226-1	457226-2	457226-3	457375	457376	457377	457378	457380	457381	457382	457383	457384	457385	457453	GT 550451	843042	872816	988670-1	988670-2	988670-7	988670-10	988670-11	988670-14	988670-16	988670-17	988670-20	988670-21	988670-22
Master oscillator	SP190013	4	12																																
Tripler	SP190004	8																																	
Transmitter master oscillator	SP190011	4		4																										4				20	
X32 multiplier	SP190023	12																			12														
X2 multiplier	SP190024	8																																	
Inverter	SP168564	4																					4	4											
Phase modulator	SP190009	4		4		4																						8							44
IF postamplifier	SP190007	4			8	16	4																									40			
TWT power supply	496174	8																																	
IF wide band limiter	SP190018	4					12																								36				
IF preamplifier, 54 mc	SP172029	4																														28			
Multiplier amplifier	SP190010	4	12																												32			16	
Input mixer	SP167565	4																																	
High level mixer	SP172382	4																																	
Beacon oscillator	SP190012	4																																	
IF amplifier	SP190022	4		16																															36
Receiver master oscillator 66.232	SP190014	4	8																								4				32				
Command receiver	475210	4						4	8	4	24														4	4		44					56		
Telemetry transmitter	475220	4										4	8	8	4	4	4	4	4	4									8		72				
Transponder regulator	475101	4																																	
Telemetry regulator	475222	4																																	
Transmitter regulator	475175	8																																	
Diplexer	475201	4																																	
Command regulator	475212	4																																	
Solar panel	475252	14																																	
Solar panel, special	475253	2																																	
Battery regulator	475251	4																				4													
Wiring harness		1																																	
Pace		4																																	
Timer and apogee driver		1																																	
Command decoder		1																																	
Jet control electronic - solenoid drive		1																																	
Telemetry encoder		1																																	
Number per spacecraft			32	24	8	20	16	4	8	4	24	4	8	8	4	4	4	4	4	4	12	4	4	4	4	4	4	52	8	4	172	68	56	36	80

MATERIAL AND PROCESSES STUDIES

The study of behavior of thermal control paints under conditions of accelerated testing is continuing.

This program has as its basis the collection of data from which long-term (3 years) behavior of thermal control coatings could be predicted. Consequently, three different types of paint systems were chosen: 1) a titanium dioxide, - epoxy, 2) antimony trioxide -potassium silicate, and 3) Hughes inorganic white. These three paint systems degrade by different mechanisms. The first and second types could be utilized inside Syncom, even though they degrade in ultraviolet.

The Hughes inorganic white was used on the exterior of Syncom I where required. It is expected that the comparison of these paints will enable a firm prediction to be made concerning the reality of accelerated testing and consequently the life expectancy of the thermal control coatings during a 3-year period.

Results to date for accelerated testing of two white coatings are shown in Figures 7-3 and 7-4. The coating of Figure 7- 3 is antimony trioxide in a potassium silicate binder. Figure 7- 4 is titanium dioxide in an epoxy binder (Skyspar untinted white, Andrew Brown Paint Company). The Skyspar paint degradation is caused by changes in both the pigment (TiO_2) and the binder (epoxy). The antimony oxide paint degradation is caused solely by the change in the pigment. There is some similarity in degradation mechanism between TiO_2 and Sb_2O_3 in that both are defect structure mechanisms. The degradation of the epoxy is caused by chain scission and/or bond rupture. When this work is done, the data will be compared with the degradation of a thermal control coating that is quite resistant to ultraviolet radiation - Hughes inorganic white - being studied on this same program as a reference.

The samples, tested in individual glass chambers evacuated by ion pumps, were irradiated at a pressure less than 10^{-8} torr. Temperature of the samples varied depending on the intensity of radiation. For 1x solar ultraviolet (defined over the region of 2200 - 4000 Å) temperatures were about 100° F, for 5x ultraviolet about 220° F, and for 10x ultraviolet 350° F for the antimony trioxide system. For the titanium dioxide-epoxy system, temperatures were: 1x - 100° F, 5x - 120° F, 10x - 300° F.

The curves show the change in solar absorptance, $\Delta \alpha_s$, plotted versus the equivalent solar hours, which are the product of the intensity level and real time. Some experiments have yet to be completed so that the curves may show some change.

In particular, the 10x curve for the antimony trioxide is inadequately defined, because these initial tests were made in the form of a grid, aimed

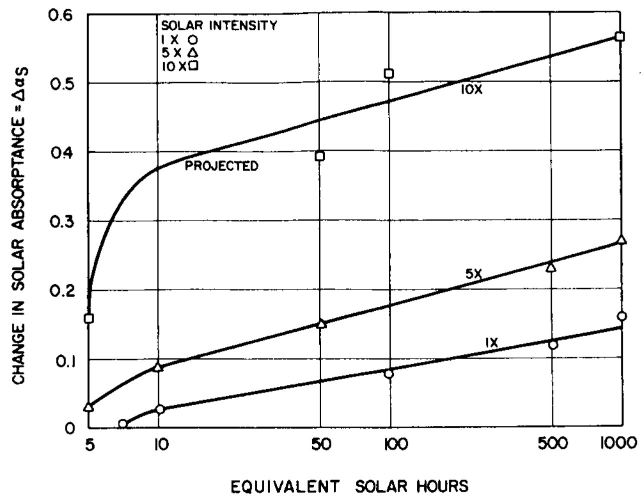


Figure 7-3. Degradation of Thermal Points

Sb_2O_2 antimony trioxide - potassium silicate

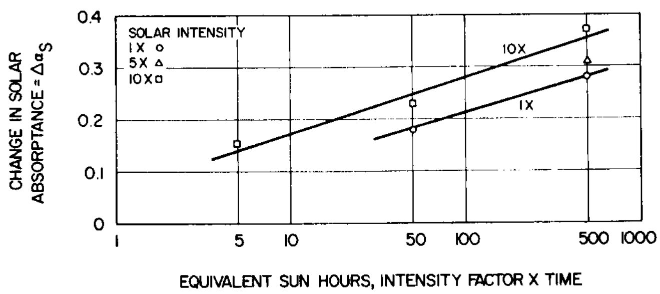


Figure 7-4. Degradation of Thermal Points

Skyspar titanium oxide-epoxy

at securing the widest possible range of information in the least possible time with available vacuum chambers. The test sequences were designed to yield long-term data by which the slope of the degradation curve could be established.

Major observations to date are:

1) Increase in solar absorptance may be defined as a logarithmic function for each intensity level.

2) At present accelerated testing without correction factors to relate intensity curves to one another does not seem possible. The nature of such factors must be determined for such tests to be quantitatively meaningful.

3) Pigment compounds in paints of varying pigment binder ratios display varying changes for the same ultraviolet radiation.

The major unresolved problem, in addition to acceleration factors, appears to be the element that temperature plays in the degradation of these coatings. Vacuum chambers in which the temperature could be kept constant under different radiation conditions were not available for these tests.

Long-term tests on the potting compounds on the TWT sections have now been running 90 days and 69 (for two sections) at collector temperatures of 204 and 214° F. The base plate temperatures on all specimens have been 172° F. All of these tests have been conducted at 10^{-7} torr. The longest test and one shorter test have not shown any visible signs of deterioration. One of the short tests has shown some degradation in the potting compound, with the cause not yet determined as it is still in the vacuum.

Consideration of possible use of beryllium in the structure has been dropped.

A decision has been reached that semiconductors will receive a basic burn in of 240 hours to eliminate parts with defects which appear quickly in operation, to be followed by a power-aging period of 90 days. It is believed that this period will disclose any tendency to drift, so that the parts with most stable behavior can be selected for flight use. Records will be kept of the behavior of individual parts, to that end. Much of this aging will be done by vendors.

Specifications for 23 types of semiconductors have been modified to provide these requirements, which have been negotiated with 11 vendors so far. Additional types will be discussed with other vendors in the near future. Advance bills of material now indicate that about 60 types of semiconductors will be used. Some of these are variations of basic family types covered by one procurement specification. Twenty-five such existing specifications and ten new ones will probably be required. Twenty-five are now qualified to the requirements of their basic specifications, unless specific requirements should be added for Syncom applications. Thirty-six will require qualification testing.

It is expected that design and manufacturing specifications, and appropriate testing to such specifications, will be required for about 50 magnetic parts.

8. SPACECRAFT SUPPORT EQUIPMENT, RELATED SYSTEMS, AND INTERFACES

PRELIMINARY INTERFACE SPECIFICATIONS

Syncom II RF and Electrical Interface Specification

1.0* SCOPE

1.1 Introduction: Under NASA Goddard Space Flight Center contract NAS5-2797, Hughes Aircraft Company is conducting feasibility studies and advanced technological development for an advanced, stationary, active repeater communications satellite. This development effort, coupled with the experience from the Syncom I program, will lead to the establishment of a stationary, active repeater communication satellite experimental program. System development requires the integration of the spacecraft, launch vehicle, and ground support equipment into an effective system.

The objectives of the experimental program will be to prove spacecraft performance and reliability and to demonstrate the feasibility of providing communication service. The development objectives of the Syncom II program will be to simplify the spacecraft design to the maximum extent possible; however, a comprehensive ground qualification and acceptance testing program is required to simulate on-orbit operation as closely as possible. In addition to integrating the spacecraft with the ground stations, provision must be made for RF and electrical interfaces with the spacecraft such that the interface connections will not degrade on-orbit operation or require spacecraft modification after qualification testing is complete.

1.2 Purpose: The Syncom II experimental flight test program will make maximum use of the existing NASA ground station and launch complex equipment and facilities. This means that the Hughes spacecraft development must be coordinated and integrated with many associated support and service organizations.

*Numbers refer to specification.

Provisions must be made for the appropriate RF, electrical, and mechanical interfaces with the spacecraft during ground testing, launch readiness, and on-orbit operations. The Syncom II RF and Electrical Interface Specification defines the spacecraft RF and electrical interfaces with the equipment and facilities required to support ground and on-orbit spacecraft testing. It will be used to coordinate development and integration of the Syncom II experimental program. The initial issue will be as complete as possible. However, amendments and added details will be included by periodic updating to reflect changes in program requirements and more detailed engineering.

2.0 APPLICABLE DOCUMENTS

1) "Initial Project Development Plan," Volume I, Technical Plan, Hughes Aircraft Company, SSD 2380R, NASA Contract 5-2797, 15 August 1962.

2) Advanced Syncom Monthly Progress Reports.

3) "S2-0100, Performance and Test Specification," Advanced Syncom Spacecraft.

4) "Pulse Frequency Modulation Telemetry Standards," GSFC Data Requirements Systems Committee, NASA-Goddard Space Flight Center, 1 November 1962.

5) "IPM PFM Encoder (S-74) (Preliminary), "Code 631.1 GSFC, NASA, Revision A, 6 August 1962.

6) "Syncom Booster Feasibility Study," Lockheed Missile and Space Company, LMSC-A057612, 30 September 1962.

7) "Syncom II Mechanical Interface Specification," 15 May 1962.

8) "Project Syncom II Procurement Specification for SSB Exciter and Power Amplifier," NASA, GSSC-S11-001, 19 December 1962.

3.0 REQUIREMENTS

3.1 Syncom II Communication Test and Control System

3.1.1 System Objectives: During launch and orbital operations the spacecraft will interface with the launch complex and the worldwide network of ground stations through RF links. A microwave (4 and 6 kmc) communication system will be used to provide communications service, and a lower frequency (VHF) control system will be used primarily for spacecraft control. Although the two independent RF links perform separate functions, their space and ground terminals must be interconnected to provide effective communication service, communication traffic control, and spacecraft control.

In addition to its primary launch and orbital functions, the ground-space communication test and control system will be used in conjunction with electrical test connections to evaluate and check out the spacecraft during ground testing operations. As a design objective a single ground system design will satisfy the Syncom II ground testing and flight requirements.

The Communication Test and Control System will satisfy the following Syncom II program requirements:

1) Synchronous orbit phase - The system will provide communication service and traffic control testing, spacecraft control, and system postflight analysis data.

2) Transfer orbit phase - The system will provide communication checkout, spacecraft control, and system postflight analysis data.

3) Parking orbit phase - The system will provide near real time spacecraft status and system postflight analysis data.

4) Prelaunch checkout - The system will provide spacecraft control, near real time spacecraft preflight checkout data, and system postflight analysis data. Provisions will be made for umbilical power during prelaunch checkout.

5) Qualification and acceptance tests - The system will be used with either electrical or RF links to control the spacecraft during testing and to provide spacecraft performance and reliability data. Provisions will be made for auxiliary power and test connections as required to adequately perform qualification and acceptance testing.

3.1.2 Communication Test and Control System Description: System design for the developmental flight test program is shown in Figures 8-1 through 8-5, as follows:

Figure 8-1	:	Synchronous orbit phase
Figure 8-2	:	Transfer orbit phase
Figure 8-3	:	Parking orbit phase

Figure 8-4 : Prelaunch

Figure 8-5 : Ground support equipment system

3.2 Communication Test System: The communication test system is made up of the spacecraft communication subsystem and the ground facilities required to provide communication system checkout and test. The Syncom II spacecraft will be able to simultaneously accommodate four independent communication links, each on its own assigned frequency. Each link will operate in a multiple access mode or in a wideband frequency translation mode.

3.2.1 Communication System Objectives: The Syncom II ground and spacecraft communication equipment will test techniques which could provide global communication service between numerous ground terminals. A system using these techniques could be integrated into the present common carrier communication net and be compatible with current voice, teletype, and television communications traffic.

The communication system will evaluate the following Syncom II spacecraft capabilities:

1) Communication traffic capacity - The spacecraft will accommodate 600 two-way voice channels or one monochrome or color television channel in each of four assigned frequency bands. The voice channels could originate from as many as 100 ground terminals simultaneously and can accommodate multiplexed teletype signals.

2) Communication quality - The quality of the communication links will meet or exceed the appropriate standards established by the International Radio Consultive Committee (CCIR) of the International Telecommunications Union (ITU).

Table 8-1 gives the development objectives of the multiple access ground/space/ground link, and Table 8-2 gives the development objectives of the wideband frequency translation ground/space/ground link. The four spacecraft transponders will be switched from the multiple access to the wideband mode by ground command as required to meet the communication service demands.

3.2.2 Developmental Orbital Operation Program Requirements: More than one ground communication terminal station will be required to demonstrate the operational prototype space vehicle performance and to evaluate total operational system feasibility. Ground support equipment

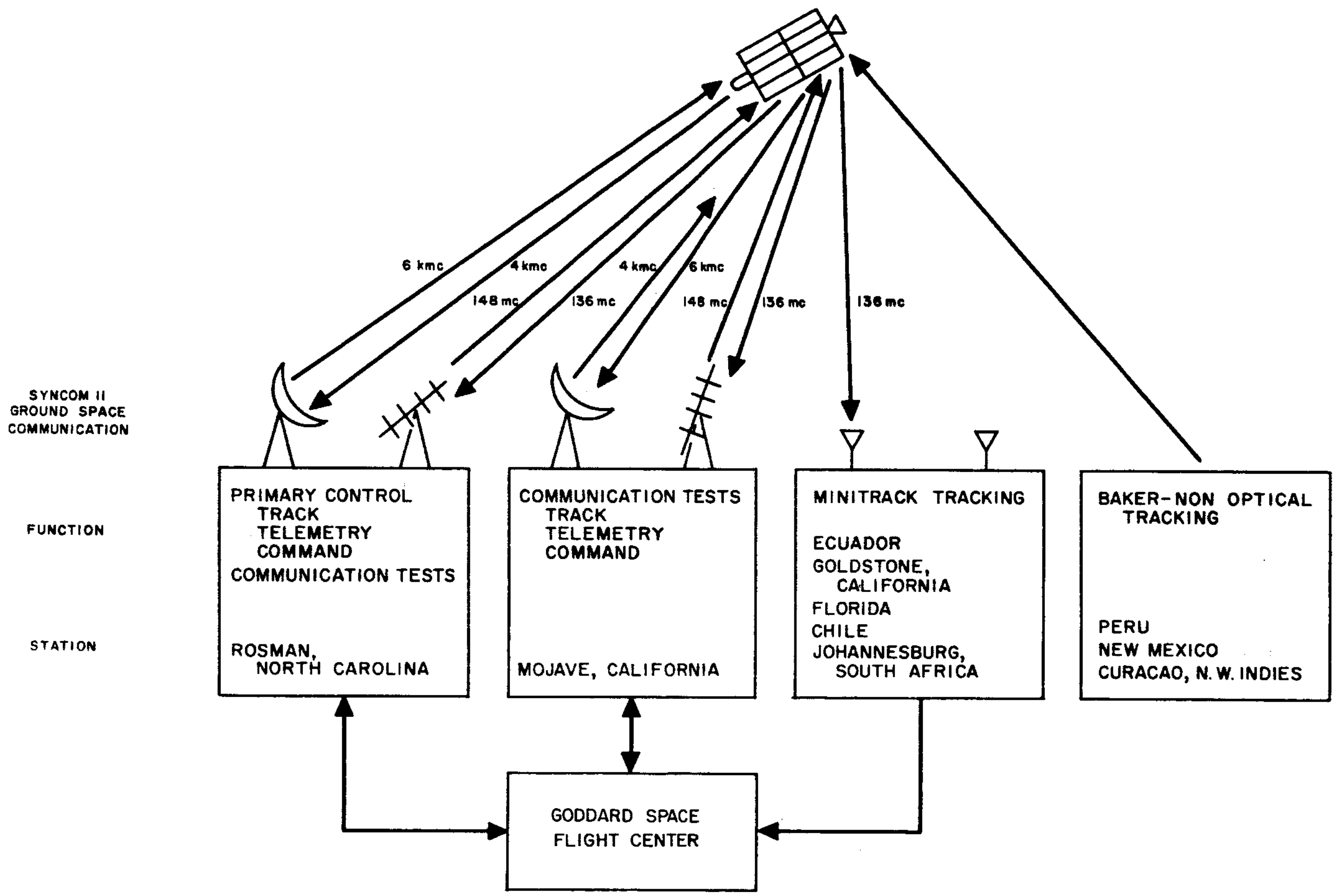


Figure 8-1. Developmental Orbital Operations
Syncom II communication test and control system (synchronous orbit phase)

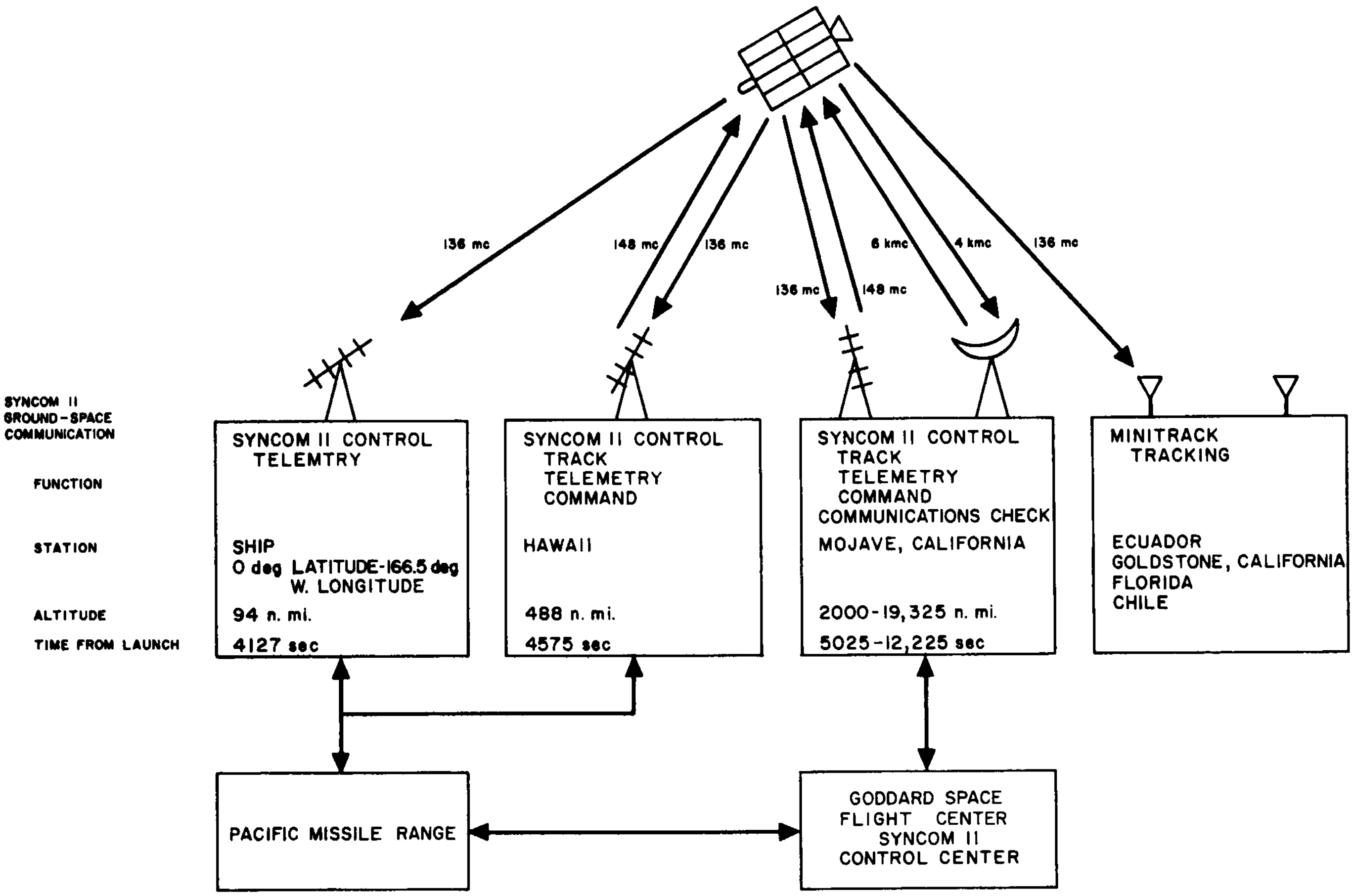


Figure 8-2. Syncom II Communication Test and Control System
Transfer orbit phase

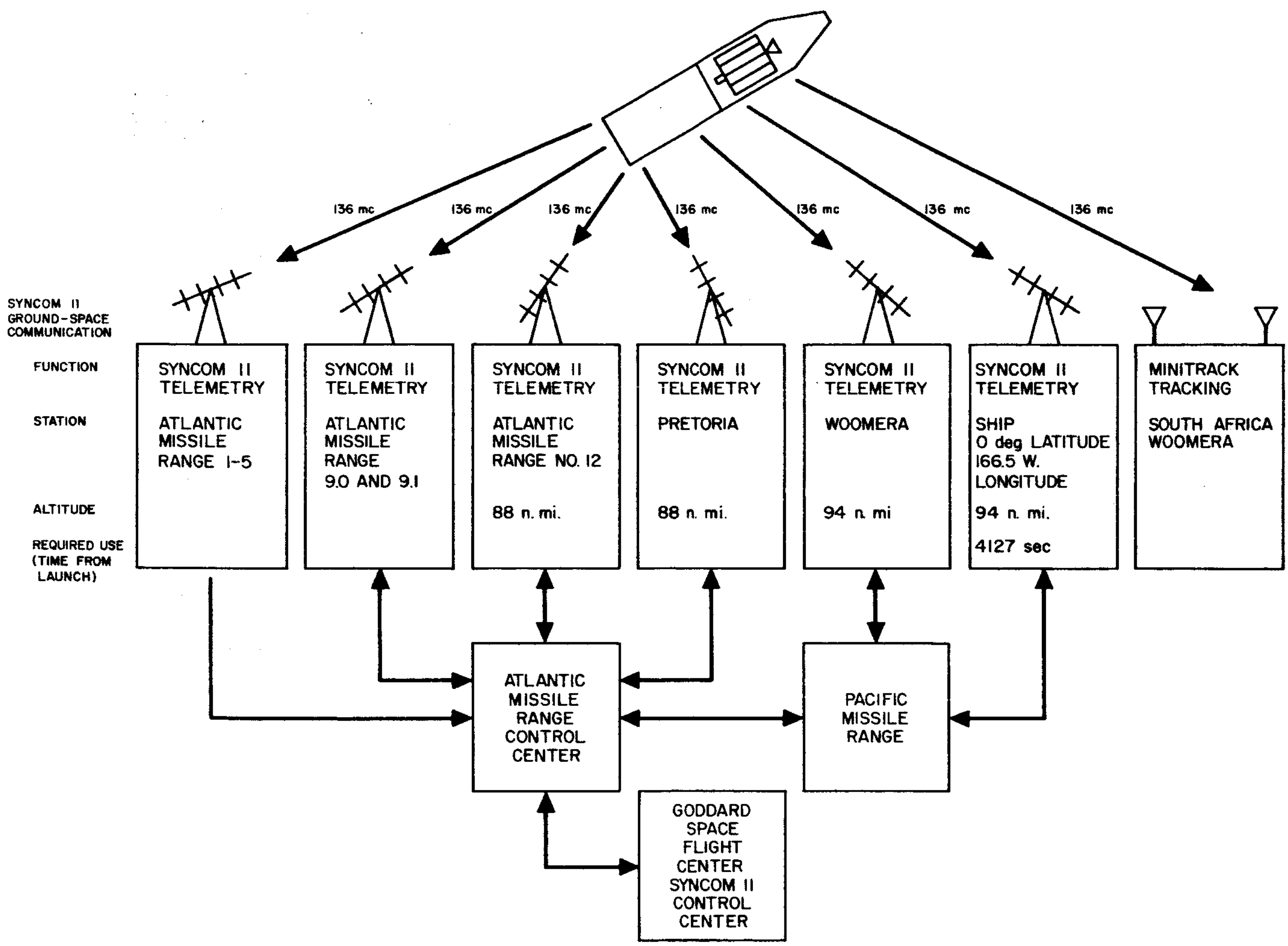


Figure 8-3. Syncom II Communication Test and Control System - Parking orbit phase

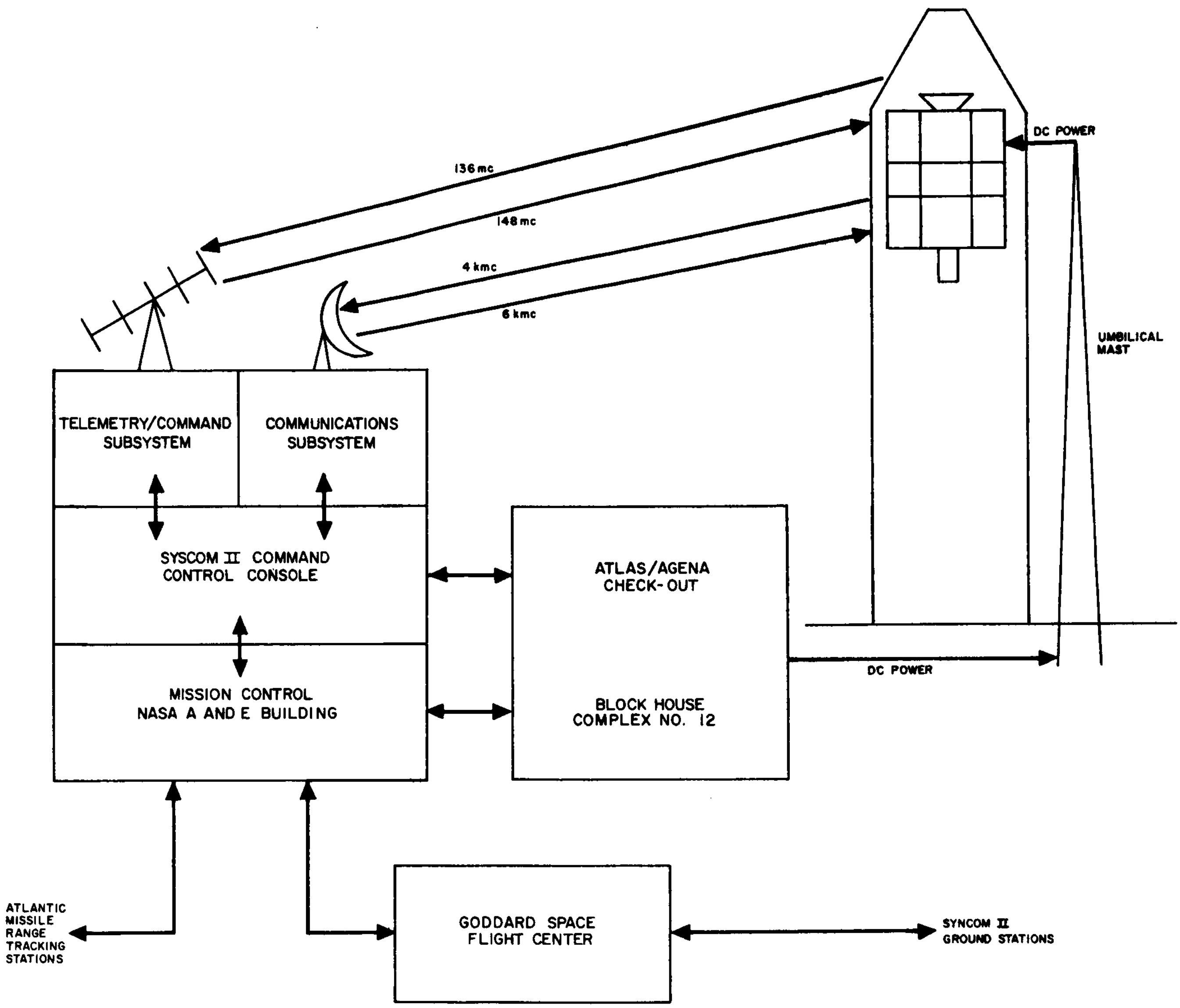


Figure 8-4. Syncom II Communication and Control System - Prelaunch

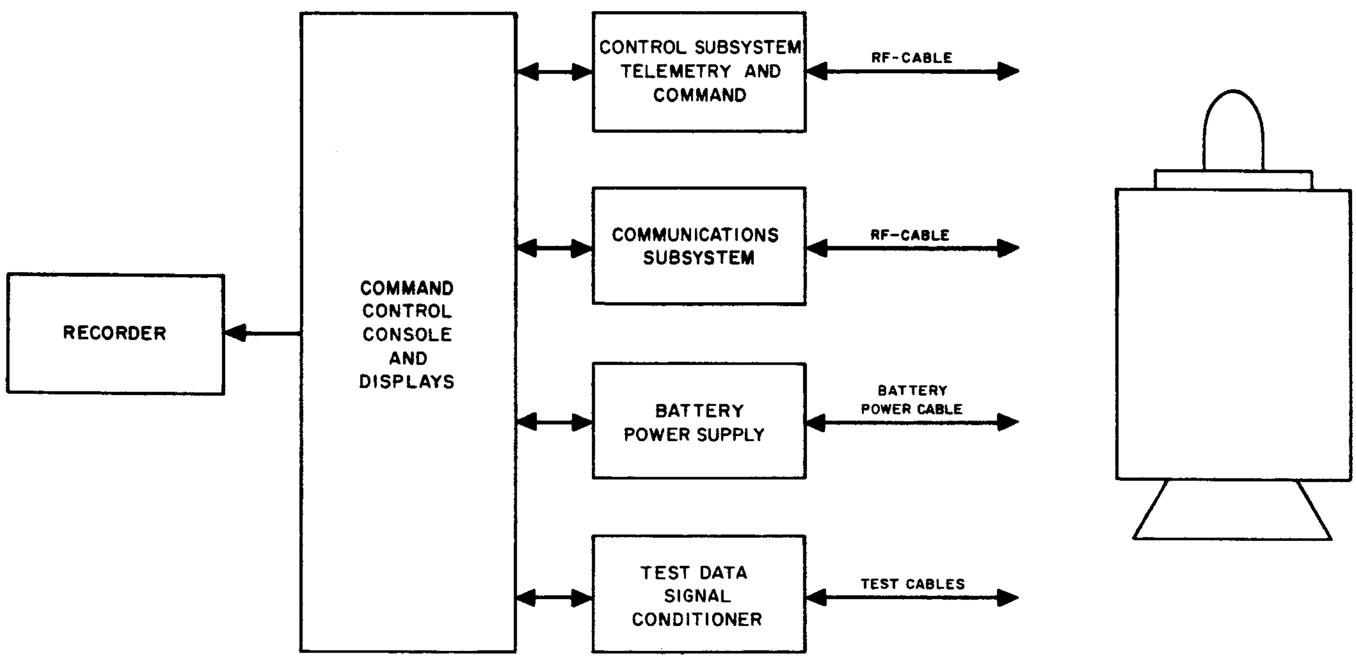


Figure 8-5. Syncom II Ground Support Equipment System

TABLE 8-1 MULTIPLE ACCESS LINK OBJECTIVES

Overall Link Capabilities	
Maximum number of links per spacecraft	4
Voice channels, maximum number per link	1200
Teletype channels, maximum number per link	18000
Frequency and level control channels per link per station	1
Tracking beacons per link	1
Overall Link Standards	
Link quality will be consistent with established CCIR standards	
Voice channels	
Test tone/noise ratio	50 db
Total channel bandwidth per voice channel	4 kc
Voice portion of bandwidth	3.1 kc
Teletype, multiplexed signals	
Maximum error in frequency	2 cps
Ground-to-Space Link Characteristics	
RF band	6000 mc
RF bandwidth required per transponder	5.8 mc
Signal characteristics	Frequency multiplexed SSB; superpressed carrier channels
Frequency stability	
Short term	One part in 10^{10}
Long term	One part in 10^{7}
Space-to-Ground Link Characteristics	
RF band	4000 mc
RF bandwidth per transponder	25 mc
Carrier deviation	±12 mc
Modulation	PM

TABLE 8-2. WIDE-BAND FREQUENCY TRANSLATION LINK OBJECTIVES

Overall Link Capabilities	
Maximum number of links per spacecraft	4
Television channels per link (monochrome or color or	1
Wide-band data channels per link	
Tracking beacons per link	1
Overall Link Standards	
Link quality will be consistent with established CCIR standards	
Television signal-to-noise ratio	
Peak-to-peak signal to weighted noise	50
Television video bandwidth	
Monochrome	4 mc
Color	4.5 mc
Ground-to-Space Link Characteristics	
RF band	6000 mc
Carrier deviation	±12.5 mc
Modulation	FM
Space-to-Ground Link Characteristics	
RF band	4000 mc
Carrier deviation	±12.5 mc
Modulation	FM

will be required to analyze both the wideband FM-FM and the multiple-access SSB-PM link performance. The wideband FM-FM link test will include level measurement, frequency response, baseband channel noise, baseband interference, TV test signals, and actual TV transmission. Multiple-access SSB-PM link tests will include multiplexing of SSB channels from two or more ground stations, baseband frequency response, baseband nonlinear distortion, channel noise, channel frequency and level control, frequency response of voice channel, channel envelope delay, voice frequency carrier, telegraph test signal, narrowband data test signal, and actual telephone conversations.

3.2.3 Communication System Description: Figure 8-6 shows the space vehicle communication subsystem design and information flow; Figure 8-7, a possible ground communication subsystem design. As stated previously, the Syncom II flight test program will use existing facilities where possible to demonstrate system feasibility.

3.2.4 Communication System RF Interface Spections (Flight Test Program):

Multiple Access Link - The specifications for the actual multiple access links to be used on the flight test program are compared with an ideal Syncom II system in Tables 8-3 through 8-6.

Wideband Frequency Translation Link - The specifications for the actual wideband links to be used on the flight test program are compared with an ideal Syncom II system in Tables 8-7 through 8-10.

3.2.5 Communication System Electrical Interface Specifications: Provisions will be made for transmitting and receiving RF communication signals to the spacecraft either by coaxial cable or RF transmission during qualification and acceptance testing. (Refer to Table 8-13.)

3.3 Spacecraft Control System: The spacecraft control system is made up of the personnel and facilities required to assess system status and to control system operation in real time during flight operation, prelaunch checkout, and ground testing. Determining system status requires the collection, transmission, collation, and analysis of specified communication link quality, telemetry, tracking, and ground station status data at specified locations on specified schedules. Commanding the space vehicle requires the encoding, transmission, and decoding of command messages at specific locations at specific times.

3.3.1 Requirements and Constraints: The Syncom II program has a particular set of control system performance requirements that must be satisfied during nominal and non-nominal space vehicle performance. The magnitude and accuracy of individual requirements vary over a wide range as the space vehicle progresses from the qualification testing through synchronous orbit operation.

TABLE 8-3. TRANSPONDER SPECIFICATIONS
(Multiple Access Mode)

	Spacecraft Quadrant			
	1	2	3	4
Transmitter				
Power, watts	4	4	4	4
Frequency, mc	3992.09	4051.08	4119.94	4178.93
Bandwidth, mc	25	25	25	25
Antenna gain, db	18	18	18	18
Diplexer and phase shifter losses, db	3	3	3	3
Tracking beacon frequencies, mc	4006.95	4066.16	4135.28	4194.49
Receiver				
Noise figure, db	9	9	9	9
Frequency, mc	6019.325	6108.275	6212.10	6301.05
Bandwidth, mc	25	25	25	25
Antenna gain, db	8	8	8	8
Losses, db	1.5	1.5	1.5	1.5

To perform real or near real time control of the space vehicle, the following capabilities must be integrated into a system:

1) Tracking and ephemeris determination

2) Telemetry

3) Command

4) Data processing and computation

5) Ground communications

In addition, the Syncom II program has a particular set of design constraints that must be satisfied:

1) Vehicle reliability

2) Vehicle weight limitations

3) Vehicle power limitations

TABLE 8-4. GROUND STATION SPECIFICATIONS

(Multiple Access Mode)

	Ideal Station	Flight Test Program Station	
		1	2
Station name		Rosman	Mojave
Station latitude			
Station longitude			
<u>Antenna</u>			
Diameter, feet	85	85	40
Efficiency (transmitter and receiver), percent	54	54	54
<u>Transmitter</u>			
Number of transmitters	4	2	2
Saturated power, kw	10	10	10
Bandwidth per channel (maximum)	5.8		
Diplexer loss, db	1	1	1
Frequency stability			
Short term	1 part in 10^{10}	1 part in 10^{10}	1 part in 10^{10}
Long term	1 part in 10^{7}	1 part in 10^{7}	1 part in 10^{7}
<u>Receiver</u>			
Number of receivers	4	2	2
Noise temperature (all sources including antenna), °K	80	85	

4) Satellite range

5) Environment conditions

6) Unintentional operation protection

7) Vehicle stability and orientation

TABLE 8-5. LINK TRAFFIC CAPABILITY

(Multiple Access Mode)

	Ideal Station	Flight Test Program Station	
		1 (Rosman)	2 (Mojave)
GROUND-TO-SPACE			
Total number of multiple access links	4	2	2
Voice channels			
Maximum number of channels per link	600	600	120
Bandwidth per channel, kc	4	4	4
Voice portion of bandwidth, kc	3.1	3.1	3.1
TTY channels			
Frequency and level control pilot tone channels/link/station	1	1	1
SPACE-TO-GROUND			
Total number of multiple access links	4	2	2
Voice channels			
Maximum number of channels per link	600	600	120
Bandwidth per channel, kc	4	4	4
Voice portion of bandwidth, kc	3.1	3.1	3.1
TTY channels			
Frequency and level control pilot tone channels/link/station	1	1	1

TABLE 8-6. LINK CALCULATIONS

(Multiple Access Mode)

	Ideal Station	Flight Test Program Station	
		Rosman	Mojave
GROUND-TO-SPACE SSB-VOICE			
Transmitter peak power capability, dbw	40	40	40
Transmitter average power, dbw	31.7	31.7	31.7
Channel test tone power, dbw	18.9	18.9	18.9
Diplex loss, db	-1.0	-1.0	-1.0
Ground antenna gain, db	62.1	62.1	55.5
Space attenuation, db	-200.8	-200.8	-200.8
Receiving antenna gain, db	8.0	8.0	8.0
Off beam center allowance, db	-1.5	-1.5	-1.5
Diplexer loss, db	-1.0	-1.0	-1.0
Received test tone power, dbw	-115.3	-115.3	-121.9
Receiver noise power density, dbw/cps	-195.3	-195.3	-195.3
Channel bandwidth, db	34.9	34.9	34.9
Psophometric noise weighting factor, db	-2.5	-2.5	-2.5
Receiver channel noise (weighted), dbw	-162.9	-162.9	-162.9
Test tone/fluctuation noise ratio, db	47.6	47.6	41.0
Test tone/intermodulation noise ratio, db	50.5	50.5	
Test tone/noise ratio, db	45.8	45.8	
SPACE-TO-GROUND PM - VOICE			
Spacecraft transmitter power, dbw	6	6	6
Diplexer and phase shifter losses, db	-3	-3	-3
Spacecraft antenna gain, db	18	18	18
Space attenuation, db	-197.1	-197.1	-197.1
Offbeam center allowance, db	-2	-2	-2
Ground antenna gain, db	58.4	58.4	51.8
Received carrier power, dbw	-119.7	-119.7	-126.3
Receiver noise power density (80°K), dbw/cps	-209.6	-209.3	
Receiver noise bandwidth (25 mc), db	74.0	74.0	

TABLE 8-6. (continued)

	Ideal Station	Flight Test Program Station	
		Rosman	Mojave
Receiver noise power (total), dbw	-135.6	-135.3	
Carrier/total noise ratio, db	15.9	15.6	
Weighted channel noise power, dbw	-177.2	-176.9	
Carrier/channel noise ratio, db	57.5	57.2	
Channel test tone modulation index	0.35	0.35	0.35
Test tone/noise ratio, db	48.5	48.2	
Compandor improvement factor overall link, db	15	15	15
Test tone/effective noise ratio, db	58.9	58.6	

TABLE 8-7. TRANSPONDER SPECIFICATIONS

(Wide-Band Frequency Translation Mode)

	Spacecraft Quadrant			
	1	2	3	4
Transmitter				
Power, watts	4	4	4	4
Frequency, mc	3992.09	4051.08	4119.94	4178.93
Bandwidth, mc	25	25	25	25
Antenna gain, db	18	18	18	18
Losses, db	3	3	3	3
Tracking beacon frequencies, mc	4006.95	4066.16	4135.28	4194.49
Receiver				
Noise figure, db	9	9	9	9
Frequency, mc	6019.325	6108.275	6212.10	6301.05
Bandwidth, mc	25	25	25	25
Antenna gain, db	8	8	8	8
Losses, db	1.5	1.5	1.5	1.5

TABLE 8-8. GROUND STATION SPECIFICATIONS

(Wide-Band Frequency Translation Mode)

	Ideal Station	Flight Test Program Station	
		1	2
Station name		Rosman	Mojave
Station latitude			
Station longitude			
Antenna			
Diameter, feet	85	85	40
Efficiency (transmit and receive), percent	54		
Transmitter			
Number of transmitters	4	2	2
Saturated power, kw	10	10	10
Bandwidth, mc	25	25	25
Diplexer loss, db	-1		
Frequency stability			
Short term	1 part in 10^{10}	1 part in 10^{10}	1 part in 10^{10}
Long term	1 part in 10^{7}	1 part in 10^{7}	1 part in 10^{7}
Receiver			
Number of receivers	4	2	2
Noise temperature (all sources including antenna), °K	80	85	

8) Station location constraints

9) Number of space programs using ground stations

10) Ground station switchover time between programs

11) Ground testing electrical test points

TABLE 8-9. LINK TRAFFIC CAPABILITY

(Wide-Band Frequency Translation Mode)

	Ideal Station	Flight Test Program Station 1 (Rosman)	Flight Test Program Station 2 (Mojave)
GROUND-TO-SPACE			
Total number of wide-band links	4	2	2
Maximum number of TV channels (monochrome)	4	2	2
Maximum number of TV channels (color)	4	2	2
Maximum number of wide-band data channels	4	2	2
SPACE-TO-GROUND			
Total number of wide-band channels	4	2	
Maximum number of TV channels (monochrome)	4	2	
Maximum number of TV channels (color)	4	2	
Maximum number of data channels	4	2	

3.3.1.1 Tracking and Ephemeris Determination Requirements: The tracking and ephemeris determination system functions are:

1) To provide accurate communication system antenna orientation;

2) To determine and schedule orbit adjust commands;

3) To schedule and control ground station operations;

4) To detect multispace program conflicts;

5) To provide ephemerides for postflight analysis.

TABLE 8-10. LINK CALCULATIONS

(Wide-Band FM Link)

	Ideal Station	Flight Test Program Station	
		Rosman	Mojave
GROUND-TO-SPACE			
Transmitter power, dbw	33.0	33.0*	40
Diplexer loss, db	-1.0	-1.0	-1.0
Ground antenna gain, db	62.1	62.1	55.5
Space attenuation, db	-200.8	-200.8	-200.8
Receiving antenna gain, db	8	8	8
Off beam allowance, db	-1.5	-1.5	-1.5
Diplexer loss, db	-1.0	-1.0	-1.0
Received carrier power, dbw	-101.2	-101.2	-100.8
Receiver noise power density, dbw/cps	-195.3	-195.3	-195.3
Receiver noise bandwidth (25 mc), db	74	74	74
Receiver noise power, dbw	-121.3	-121.3	-121.3
Carrier/noise ratio, db	20.1	20.1	20.5
SPACE-TO-GROUND			
Spacecraft transmitter power, dbw	6	6	6
Diplexer and phase shifter losses, db	-3	-3	-3
Spacecraft antenna gain, db	18	18	18
Space attenuation, db	197.1	197.1	197.1
Off beam center allowance, db	-2	-2	-2
Ground antenna gain, db	58.4	58.4	51.8
Received carrier power, dbw	-119.7	-119.7	-126.3
Receiver noise power density (80°K), dbw/cps	209.6	209.3	
Receiver bandwidth (25 mc), db	74.0	74.0	
Receiver noise power, dbw	135.6	135.3	
Carrier/noise ratio, db	15.9	15.6	
Carrier/noise ratio - up link, db	20.1	20.1	20.5
Carrier/total noise ratio, db	14.5	14.5	
Top modulation frequency, mc	4	4	

*Assumes only 2 kw is radiated although maximum transmitter capacity is 10 kw.

TABLE 8-10. (continued)

	Ideal Station	Flight Test Program Station	
		Rosman	Mojave
Modulation index, M	2.5	2.5	
Improvement factor, $3M^2 \frac{25}{4(2)}$, db	17.7	17.7	
Average signal-to-noise ratio, db	32.2	32.2	
Noise weighting factor, db	14	14	14
Peak-to-peak signal/weighted noise, db	55.2	55.2	

3.3.1.2 Space Vehicle Telemetry System Requirements: The spacecraft telemetry system functions are to provide:

1) Prelaunch checkout data;

2) Spacecraft status required for nominal and non-nominal on-orbit control;

3) Data for postflight evaluation of non-nominal launch; injection, or in-orbit spacecraft performance;

4) Qualification and acceptance test data.

3.3.1.3 Space Vehicle Command System Requirements: The command system functions are:

1) To control spacecraft operation during prelaunch checkout;

2) To provide backup command to start apogee injection motor;

3) To control spacecraft in-orbit operation;

4) To control spacecraft during qualification and acceptance testing.

3.3.1.4 Electrical Test Cable Requirements: Spacecraft electrical connection to the spacecraft are required to provide:

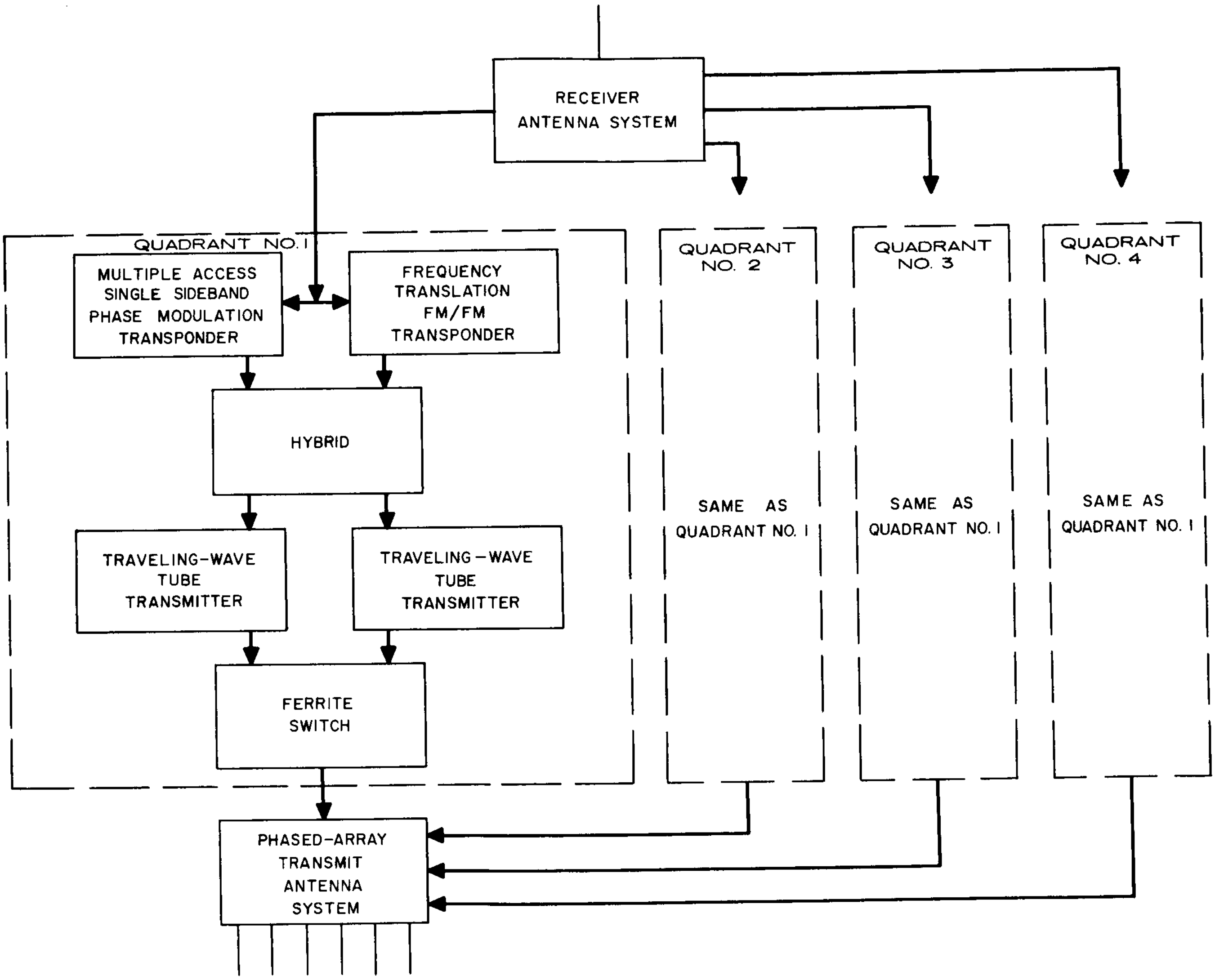


Figure 8-6. Syncom II Spacecraft Communication Subsystem

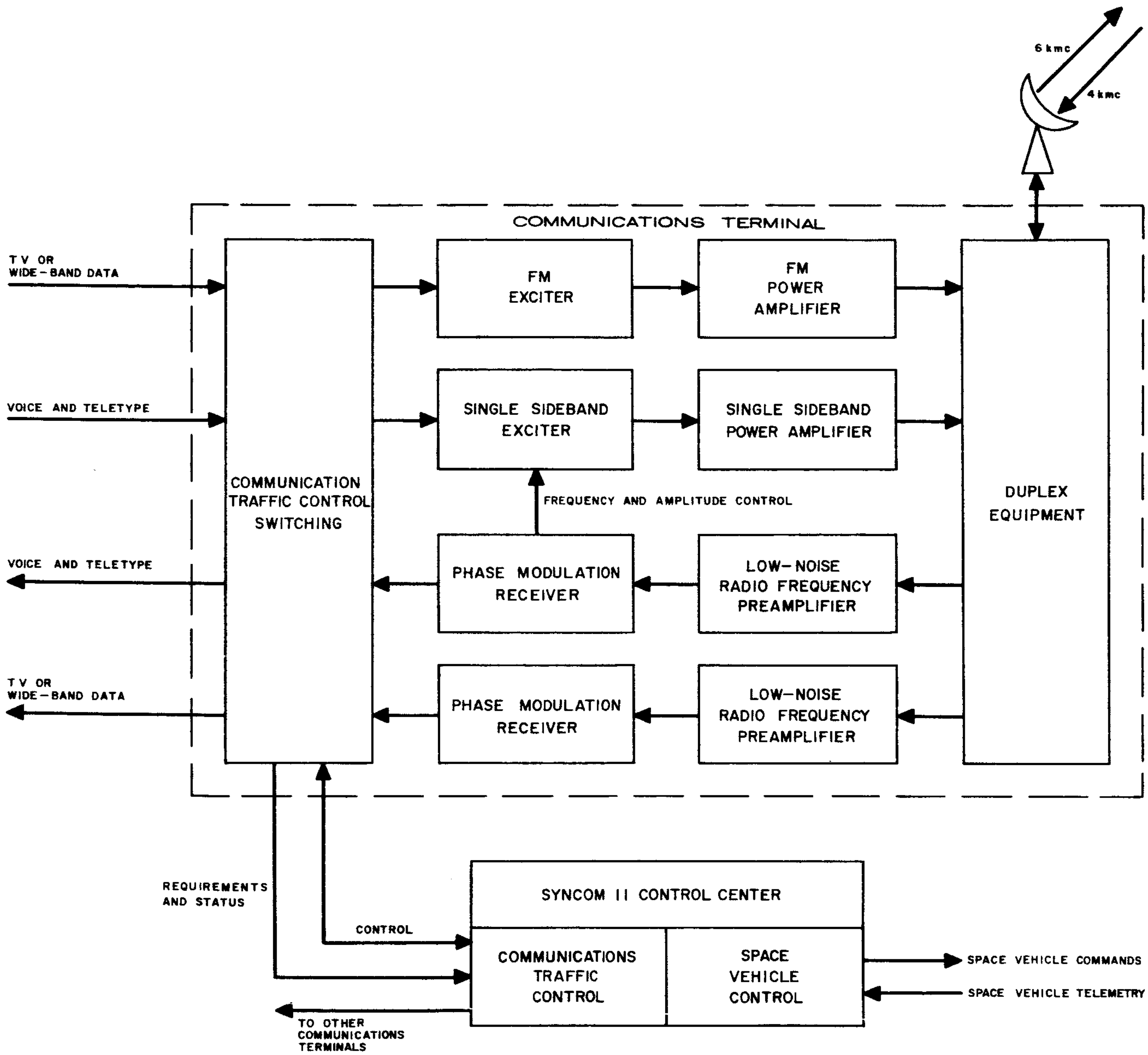


Figure 8-7. Possible Syncom II Ground Communication Subsystem

1) Qualification and acceptance test data not available from the spacecraft telemetry or communication subsystems;

2) Spacecraft battery power for ground testing and prelaunch checkout;

3) Transmission of T/M and command RF signals during qualification and acceptance testing.

3.3.2 Control System Description: The primary objectives in selecting a control system design were to simplify the overall spacecraft system and maximize spacecraft reliability. Therefore, a single spacecraft control system has been selected to perform ground testing, prelaunch checkout, launch, and injection status, and in-orbit control functions. The control system will include a modified Goddard standard telemetry subsystem, modified Goddard standard FSK command subsystem, and Minitrack tracking subsystem. Baker-Nunn optical tracking will augment the radio tracking data for fine synchronous orbit adjustment.

The spacecraft telemetry subsystem is the primary means of assessing spacecraft performance for prelaunch checkout, in-orbit control and postflight analysis. Figure 8-8 shows the telemetry subsystem design and information flow for Syncom II spacecraft and Figure 8-9, that for the Syncom II ground net during prelaunch, launch and injection, and on-orbit operations.

The spacecraft command subsystem includes the personnel and facilities required to control, encode, transmit, and decode ground-to-space commands. As a design objective the command subsystem will provide complete launch pad (prelaunch) checkout control. Figure 8-10 shows the command subsystem design for the spacecraft, and Figure 8-11 shows the command subsystem design for prelaunch and on-orbit ground network.

3.3.3 Control System RF Interface Specifications

3.3.3.1 Tracking Subsystem Specifications: The Agena tracking system will provide accurate tracking and ephemeris data up to the time of Agena separation. Then ephemerides will be calculated from range/range rate and Minitrack cosine angle tracking data and acquisition will be accomplished with the self-tracking communication antennas. Baker-Nunn optical tracking will augment the radio tracking data for five synchronous orbit adjust.

3.3.3.2 Telemetry Subsystem Specifications: The system parameters of the telemetry encoder have not yet been firmly specified. Preliminary parameters are as follows:

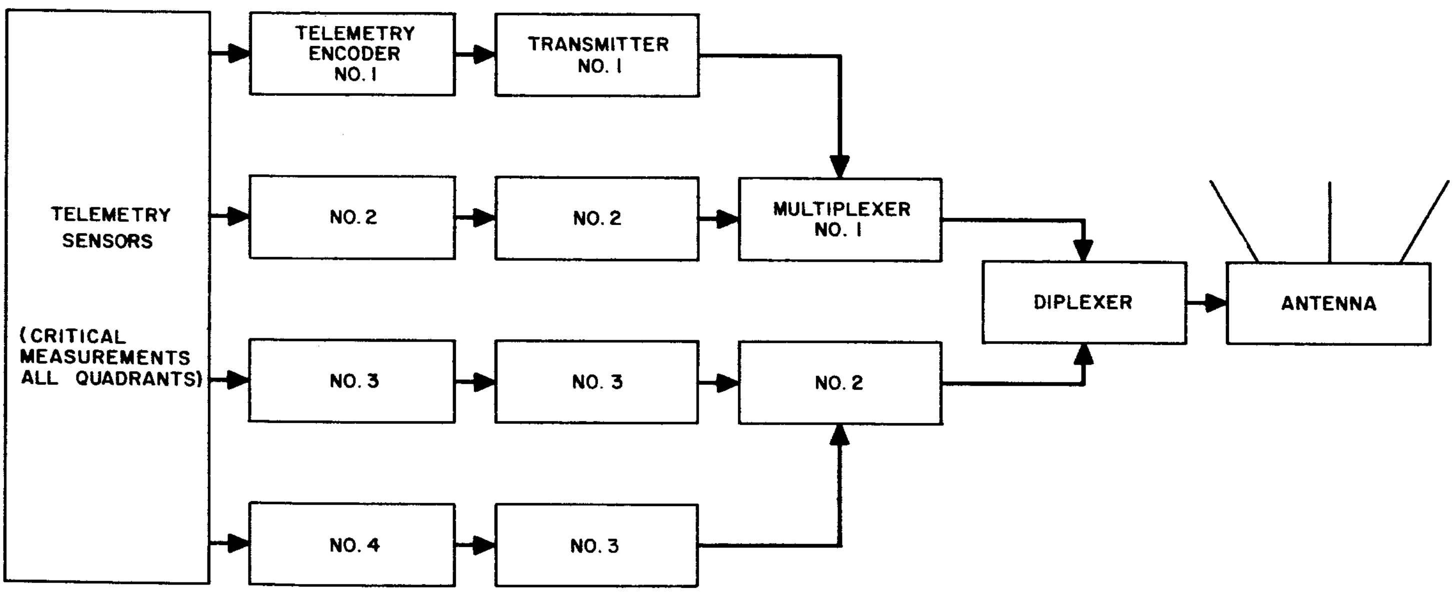


Figure 8-8. Syncom II Telemetry Subsystem
Space vehicle

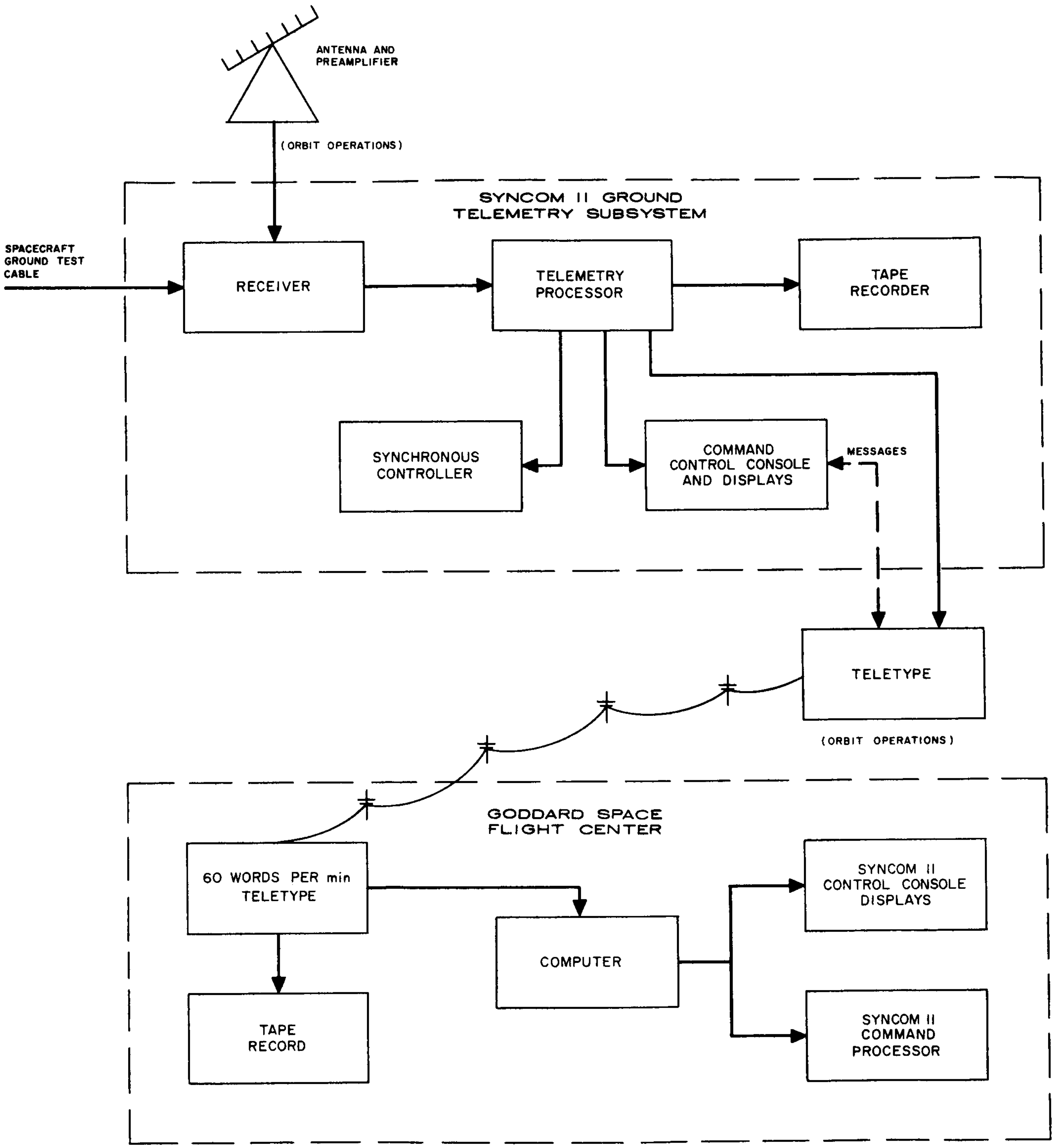


Figure 8-9. Syncom II Telemetry Subsystem
Ground station for on-orbit control

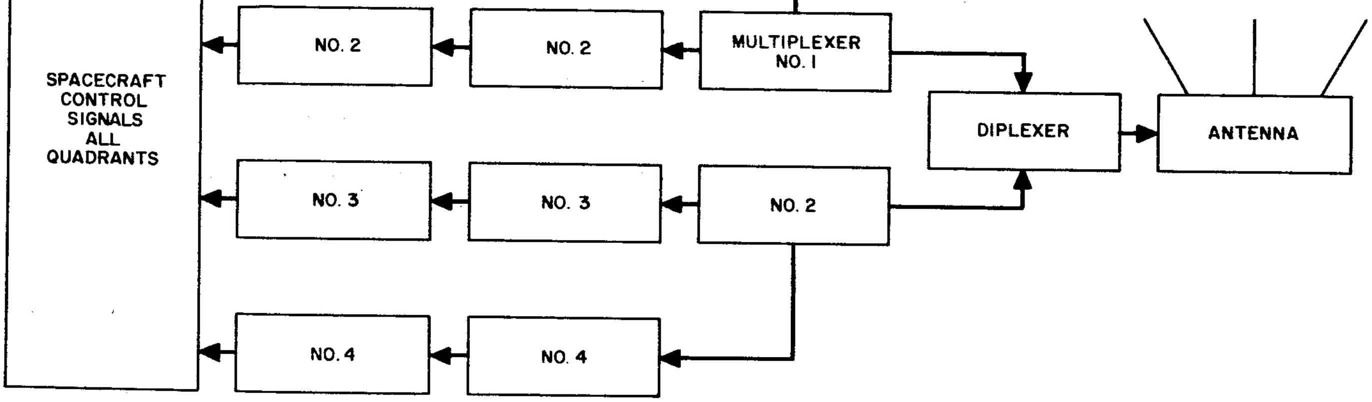


Figure 8-10. Syncom II Command Subsystem
Space vehicle

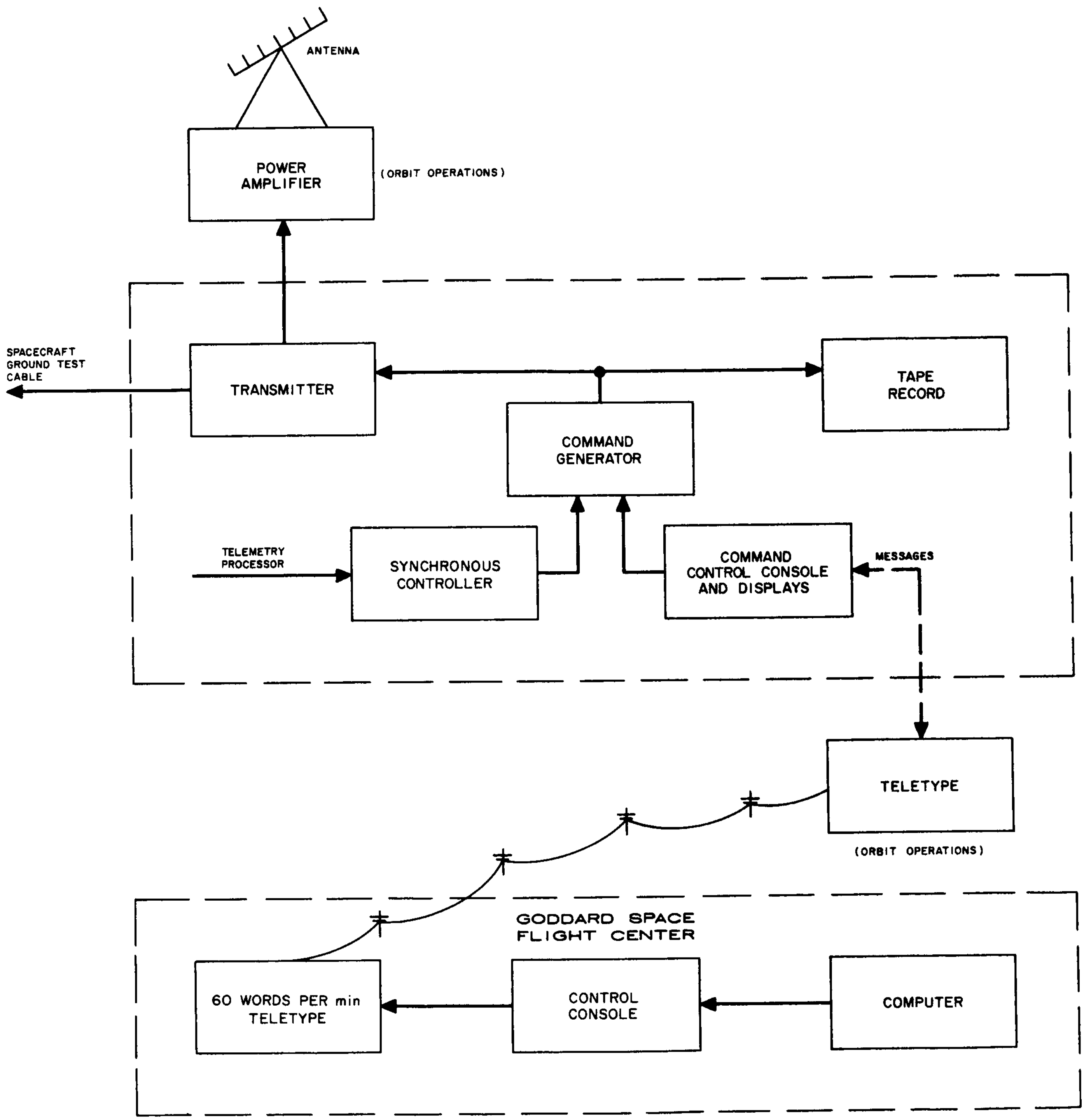


Figure 8-11. Syncom II Command Subsystem
Ground station for on-orbit control

RF band	136 mc
Modulation	PFM
Subcarrier	14.5 kc ±7.5 percent
Input voltage	0 to -4 volts
Input load	Less than 60 microamperes
Data accuracy	Encoder error less than 1 percent
Number of channels	64
Channel rate	8 per second, ±5 percent
Command execute	Command execute tone replaces subcarrier for duration of tone
Solenoid operation	Command execute tone reduced in amplitude for duration of control pulse
Pulses	Directly modulate telemetry carrier

Telemetry Link Calculations:

Transmitter power	To be determined
Cable losses	0.13 db
Attenuator loss	1.50 db
Diplexer loss	0.70 db
Hybrid-balun loss	0.50 db
Antenna gain (worst case)	-3.20 db
Free space loss	167.40 db
Ground terminal	Parameters to be determined after selection of ground stations

Flight Test Program Telemetry Readout Requirements:

See Table 8-11

3.3.3.3 Command Subsystem Specifications: The command subsystem is designed in accordance with the NASA-Goddard Space Flight Center command system guidelines to permit maximum utilization of existing NASA ground facilities.

RF Link Characteristics:

Frequency	148.260 mc
Modulation	FSK
Frequency stability	1.0×10^5

Command information channels

Frequency-shift-keyed tone modulation channels

Execute tone

Zero tone

One tone

Amplitude modulating of one or zero tone channel bit synchronization - one bit per sine wave

Flight Test Program Command Requirements:

See Table 8-12

3.3.4 Control System Electrical Interface Specifications

3.3.4.1 Prelaunch Umbilical Connections: External power will be supplied to the spacecraft unregulated power bus through the umbilical during prelaunch checkout activities(see Table 8-13). The power cable will be fed through the nose shroud by two three-pin electrical connectors in series. They will permit umbilical release from the launch tower and disconnect from the spacecraft at shroud separation.

3.3.4.2 Qualification Test Electrical Connections: Cable requirements are included in Table 8-13.

3.3.4.3 Acceptance Test Electrical Connections: Cable requirements are included in Table 8-13.

TABLE 8-11. SYNCOM II TELEMETRY READOUT REQUIREMENTS

Measurement	Form	Channels	DATA USE					
			Control Center	Synchronous Orbit Stations	Transfer Orbit Stations	Parking Orbit Stations	Prelaunch Checkout	Qualification & Acceptance Testing
Power								
Unregulated bus voltage Battery voltage Solar panel temperature	Analog Analog Analog	2 2 1	Display Analysis	Display Record TTY	Display Record TTY	Record TTY*	Display Record	Display Record
Pace:								
FLL lock-timer selection Antenna beam position	Digital Digital	1 3	Display Analysis	Display Record TTY	Display Record TTY	Record	Display Record	Display Record
Command								
Command verification	Digital	3		Display Record	Display Record		Display Record	Display
Control								
ψ_2 angle Propellant tank pressures Propellant tank temperatures	Digital Analog Analog	3 4 2	Display Analysis	Analysis Display Record TTY	Analysis Display Record TTY	Record TTY*	Display Record	Analysis Display Record
Transponder								
Transmitter power Receiver signal strength Receiver mode - TWT selection	Analog Analog Digital	4 4 1	Display Analysis	Display Record TTY	Display Record TTY		Display Record	Display Record
Miscellaneous								
Spacecraft identification Telemetry quadrant identification Sync-calibrate Telemetry radiated power Temperatures Radiation experiment	Digital Digital Analog Analog Analog Analog	1 1 2 1 2 5	Display	Display Record TTY	Display Record TTY	Record TTY*	Display Record	Display Record
Total		42						
Spares		22						
Total Channels		64						

*Ship station only (Agena/spacecraft separation data)

Display - Real time processing for quick-look analysis
Analysis - Real time electronic analysis
Record - Permanent record transported to control center
TTY - Real or near real time transmission to control center

TABLE 8-12. SYNCOM II COMMAND REQUIREMENTS

Command Function	COMMAND STRUCTURE		Number of Commands	GROUND STATION COMMAND SYSTEM FUNCTIONS					
	Address Bits	Magnitude Bits		Control Center	Synchronous Orbit Stations	Transfer Orbit Stations*	Parking Orbit Stations	Prelaunch Checkout	Qualification & Acceptance Testing
Command execute	CW tone				Control Encode Execute	Control Encode Execute		Control Encode Execute	Control Encode Execute
Transponder power Multiple access - ON Frequency translation - ON			 4 4	Control	Encode Execute	Encode + Execute +		Control Encode Execute	Control Encode Execute
TWT power TWT filaments - ON TWT high voltage - ON TWT and transponder power Quadrant - OFF			 8 8 4	Control	Encode Execute	Encode + Execute +		Control Encode Execute	Control Encode Execute
Central timer select			4	Control	Encode Execute	Encode Execute		Control Encode Execute	Control Encode Execute
Jet fire backup command			16	Control	Encode Execute				Control Encode Execute
Jet fire angle			4	Control	Encode Execute				Control Encode Execute
Antenna beam angle				Control	Encode Execute	Encode + Execute +		Control Encode Execute	Control Encode Execute
Telemetry power on-off			8	Control	Encode Execute	Encode Execute		Control Encode Execute	Control Encode Execute

* No ship station command requirements

\+ Mojave, California, station only

Control - Schedule and select commands
Encode - Encode command message format
Execute - Transmit command to spacecraft

TABLE 8-13. SPACECRAFT ELECTRICAL CONNECTIONS

Function of Cable	Number of Cables	Spacecraft Connection Point	CABLE USES		
			Prelaunch Checkout	Qualification Testing	Acceptance Testing
Spacecraft DC power	1	Unregulated power bus	Umbilical spacecraft power	Spacecraft power	Spacecraft power
Communication transponders RF power output	4 (one each quadrant)	Directional couplers between ferrite switch and multiplexer		RF test signal input	RF test signal input
RF power input	1	Directional coupler between antenna and multiplexer		RF test signal output	RF test signal output
Telemetry/Command RF power output RF power input	4 (one each quadrant)	Directional coupler between diplexer and balm		Spacecraft control and data readout	Spacecraft control and data readout
PACE Electronics VCO and F-100 output	4 (one each quadrant)			Check VCO frequency and F-100 pulse spacing	Check VCO frequency and F-100 pulse spacing
Input to apogee timer	4 (one each quadrant)			Control apogee timer	Control apogee timer
Propulsion subsystem Input to squib	4 (one each quadrant)			Test squib firing circuit	Test squib firing circuit

Syncom II Mechanical Interface Specification

1.0 SCOPE

1.1 Introduction: Under NASA Goddard Space Flight Center contract NAS 5-2797, Hughes is conducting feasibility studies and advance technological development for an advanced, stationary, active repeater communications satellite.

This development effort coupled with the experience from the Syncom I program will lead to the establishment of a stationary, active repeater communication satellite experimental program. System development requires the integration of the spacecraft, launch vehicle, and ground support equipment.

The Syncom II spacecraft will be launched and injected into a highly elliptical transfer ellipse by the flight-proven Atlas/Agena launch vehicle system. The standard Agena-D has successfully performed a wide variety of space missions. This has been accomplished by developing a basic Agena-D plus a variety of standard optional off-the-shelf add-on kits to perform specific functions for using programs.

The Syncom II launch vehicle system includes the Atlas-D first-stage booster, the Agena-D intermediate-stage booster, and the Syncom II applicable spacecraft support system. The Agena-D is made up of the basic standard Agena plus standard optional equipment designed for Agena-D use. The Syncom II peculiar spacecraft support system is made up of the spacecraft Agena-D adapter, the spin table, the spacecraft separation system, and the nose shroud. Figure 8-12 shows the general Syncom II prelaunch configuration.

1.2 Purpose: The Syncom II experimental flight test program will make maximum use of the flight-proven techniques and hardware of the standard Atlas/Agena launch vehicle system and existing launch complex equipment and facilities. This means that the Hughes spacecraft development must be coordinated and integrated with many associate support and service organizations.

Provisions must be made for the appropriate RF, electrical, and mechanical interfaces with the spacecraft during ground testing, launch readiness, and on-station operations. The Syncom II Mechanical Interface Specification defines the spacecraft mechanical interfaces with the equipment and facilities required to support, handle, service, and ship the spacecraft. The RF and electrical interfaces are defined in the Syncom II RF and Electrical Interface Specification.

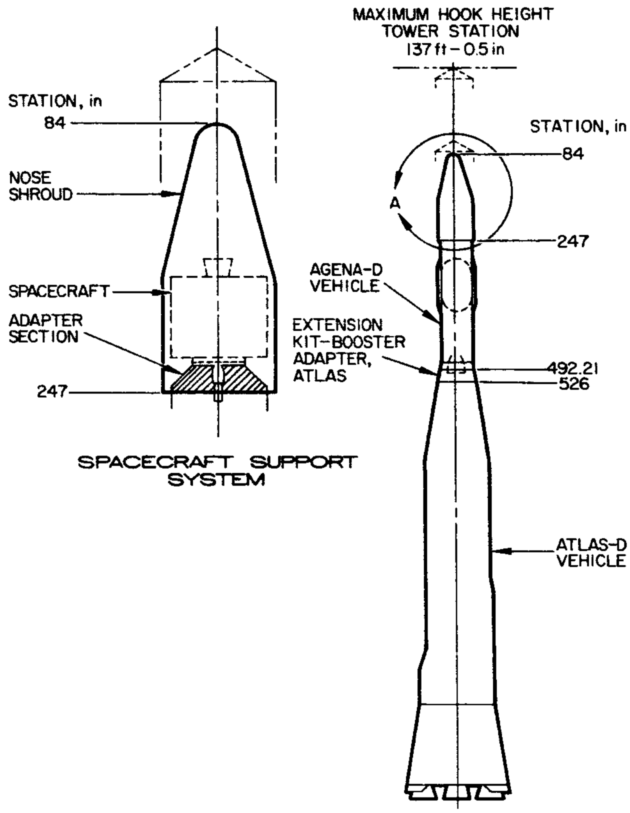


Figure 8-12. Atlas D/Agena D/Spacecraft Combination General Arrangement

This specification will be used to coordinate development and integration of the Syncom II Program. The initial specification will be as complete as possible; however, ammendments and/or added details will be included by periodic updating to reflect changes in program requirements and more detailed engineering.

2.0 APPLICABLE DOCUMENTS

2.1 LMSC Report No. LMSC-A057612: "Syncom Booster Feasibility Study Final Report", 30 September 1962 (confidential).

2.2 TWX - A.E. Jones, GSFC: "Official GSFC Vibration Requirements", 15 April 1963.

3.0 REQUIREMENTS

3.1 Launch Vehicle Interfaces

3.1.1 Launch Vehicle Performance Objectives: The launch vehicle will supply the boost and guidance necessary to separate the spacecraft on a transfer ellipse in a condition that will permit the spacecraft to complete the Syncom II mission objectives. The launch vehicle system will provide for nose shroud ejection, spacecraft apogee thrust vector orientation, spacecraft spinup, and mission support tracking and telemetry data prior to spacecraft separation. The launch vehicle system will be configured to permit adequate accessibility and RF transparency for the prelaunch spacecraft checkout activities.

3.1.1.1 Spacecraft Separation Ephemeris: The Agena SS/D timer will program a second Agena burn at the second node of the parking orbit to put the spacecraft into a highly elliptical Hohman transfer orbit having an apogee at or near the synchronous orbit altitude.

Parking Orbit Parameters

Velocity	25,620 fps (4.2.2 n. mi./sec)
Period	5270 seconds (1.468 hours)
Longitude shift eastward per circuit	338 degrees
Radius	Approx. 3530.2 nautical miles

Transfer Orbit Parameters

Inclination	29.1 degrees
Semimajor axis	13,141.25 nautical miles
Period	27.800 seconds (10.51 hours)
Apogee velocity	5250 fps (0.863 n. mi./sec)
Longitude shift eastward per circuit	202.2 degrees

3.1.1.2 Nose Shroud Ejection: The Agena SS/D timer will program ejection of the nose shroud during the first Agena burn.

3.1.1.3 Apogee Thrust Vector Orientation: After the Agena second burn, the Agena SS/D timer will program a 53-degree yaw turn to the right of the flight path in a horizontal plane to pre-align the spacecraft apogee motor thrust axis.

3.1.1.4 Spacecraft Spinup: The Agena SS/D timer will program the launch vehicle to spinup the spacecraft to a nominal 100 rpm after the apogee thrust vector has been aligned. Spin rate will be accurate to ± 10 rpm.

3.1.1.5 Launch Vehicle Support Data Requirements: The launch vehicle will provide tracking and telemetry data required to support spacecraft on-orbit control operations and postflight analysis activities.

3.1.1.5.1 Agena Tracking Data: FPS-16 radars in the ground-based communications and control network will be used to track the launch vehicle C-band beacon from launch to spacecraft separation. The Agena tracking and engine burn telemetry data will be used to provide near real time ephemerides for use as a point of departure for spacecraft tracking system. Agena ephemeris measurement accuracy requirements (to be determined).

3.1.1.5.2 Agena Telemetry Data: The launch vehicle telemetry system will provide information such as spin axis orientation, spin rate, time of second burn, and separation conditions required to support real-time spacecraft control. In addition, it will provide a spacecraft/launch vehicle interface environmental time history for postflight analysis.

3.1.1.5.3 Agena/Spacecraft Interface Support Data Requirements: Measurement accuracy, time resolution and data distribution (to be determined).

3.1.1.6 Prelaunch Spacecraft Access Requirements: Provision will be made for limited access to the spacecraft during the prelaunch checkout activities while the spacecraft is mated to the launch vehicle.

3.1.1.6.1 Adapter Cone Access: Access to the spacecraft communication antenna will not be required after the spacecraft is mounted on the Agena vehicle.

3.1.1.6.2 Nose Cone Shroud Access: An 8- to 10-inch hand access hole will be required for installation of the umbilical power, the battery disconnect and the reaction jet and apogee motor squib enabling plugs. The access hole will be located at approximately LMSC station number 180.

3.1.1.7 RF Window Requirements: RF transmission will be required through the spacecraft adapter to check out the spacecraft communication system during prelaunch activities. The adapter cone panels will be constructed of 1/8-inch-thick laminated No. 143 glass fabric with phenolic resin for this purpose. If communication checks are to be performed after launch but before spacecraft separation the antenna pattern will have to be evaluated.

3.1.2 Spacecraft Launch Vehicle Interface Description: The Syncom II-peculiar spacecraft support system will be mounted on the forward end of the Agena-D. It is made up of the spacecraft adapter section and the nose shroud. The adapter section will be a truncated cone stiffened by eight longerons and two machined end rings. It will be designed to provide structural support for the Syncom II spacecraft during the prelaunch, launch, and parking orbit phases of the flight and will include the spin and separation systems. The shroud will be attached to the basic Agena and will provide umbilical connections for prelaunch spacecraft power and air-conditioning. Figure 8-13 shows a general description of the spacecraft support system in the prelaunch position.

3.1.2.1 Adapter/Spacecraft Interface: The spacecraft will have a mechanical interface with the Agena-D adapter section, but no electrical interface is required. The spacecraft will mate at the adapter/spacecraft interface ring which has a 31-inch outside diameter. The joint is secured by a V-band clamp assembly consisting of three spring steel bands connected by three double-ended, dual bridge-wide explosive bolts. The spacecraft communication antenna housing will fit into a 24-inch-diameter well in the top end of the Agena, and the communication antenna will radiate through the RF windows during prelaunch communication system checkout. Figure 8-14 shows the spacecraft/Agena adapter section interface and Figure 8-15 shows the details of the separation joint.

Separation switches on the adapter section will be depressed by the spacecraft/adapter separation joint so that they will go to normal position at spacecraft separation. Two pairs of redundant switches

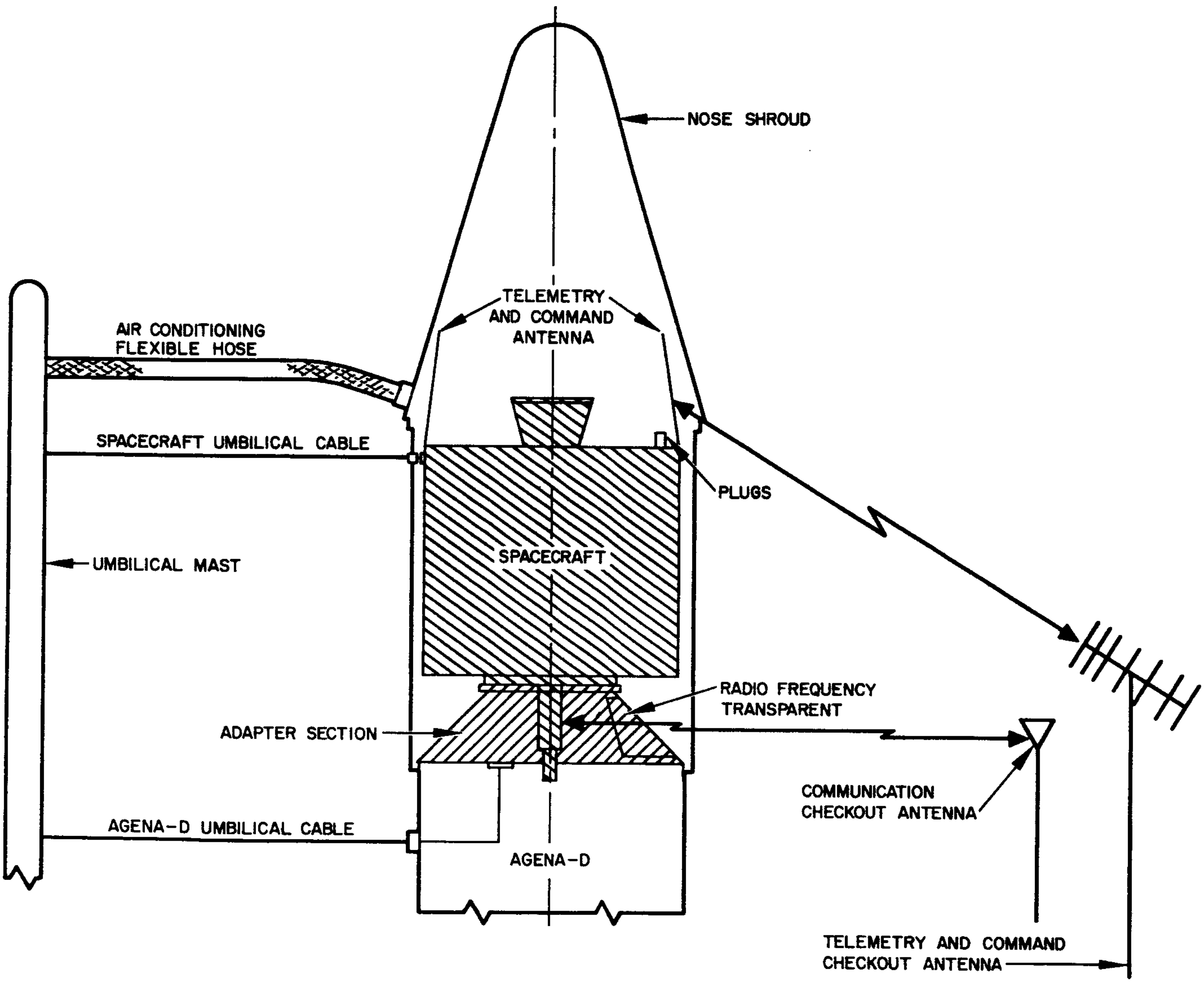


Figure 8-13. Prelaunch Spacecraft Support System Configuration

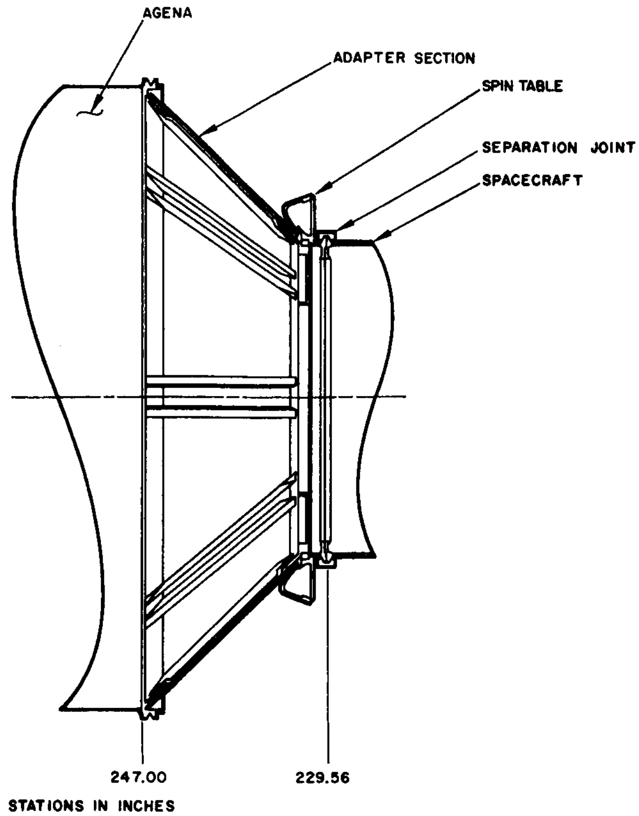


Figure 8-14. Spacecraft/Agena Interface

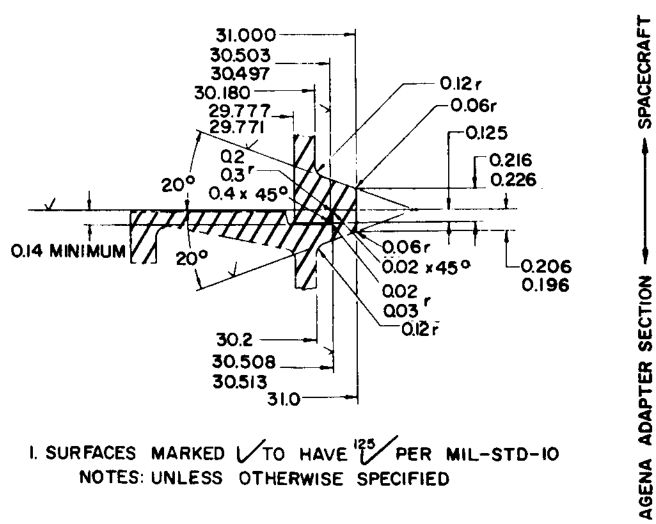


Figure 8-15. Spacecraft Agena Adapter Separation Joint

will be required to detect and transmit the separation time to the ground station by Agena telemetry.

3.1.2.2 Nose Shroud/Spacecraft Interface: An umbilical hose into the shroud will provide air-conditioning for the spacecraft during prelaunch activities, but the only interface between the spacecraft and the shroud is an umbilical cable to provide power for prelaunch ground checkout of the spacecraft. Two three-pin electrical connectors in series will permit umbilical release from the launch tower and disconnect from the spacecraft at shroud separation.

The shroud is a shortened version of the Douglas Aircraft Company Nimbus design, and is made of 91-LD phenolic fiberglass laminated skin with aluminum alloy stiffening rings. The shroud separates in clamshell fashion by segmenting two bands that hold the halves of the shroud together. A microquartz insulation blanket on the inner surface of the skin provides thermal protection to the spacecraft during ascent. Figure 8-16 shows the configuration of the shroud. Figure 8-17 shows the spacecraft envelope requirements.

3.1.3 Spacecraft/Launch Vehicle Interface Specification

3.1.3.1 Vibration Levels for Evaluation of Advanced Syncom Structure: To be applied at the separation plane, sinusoidal excitation, three axes, logarithmetic sweep at two octaves per minute, 4.35 minutes duration.

5 - 15 cps	0.25-inch double amplitude
15 - 250 cps	3 g peak
250 - 400 cps	5 g peak
400 - 2000 cps	7.5 g peak

Random excitation, three axes, 6 minutes duration per axis

20 - 80 cps	0.04 g^2/cps
80 - 1280 cps	Increasing from 0.04 g^2/cps at 1.22 db per octave
1280 - 2000 cps	0.07 g^2/cps

Exception to the above may be taken where the predominate longitudinal and lateral frequencies are sufficiently decoupled from those of the Atlas/Agena with the spacecraft attached. In this case the spacecraft response at the center of gravity may be limited to that of the separation

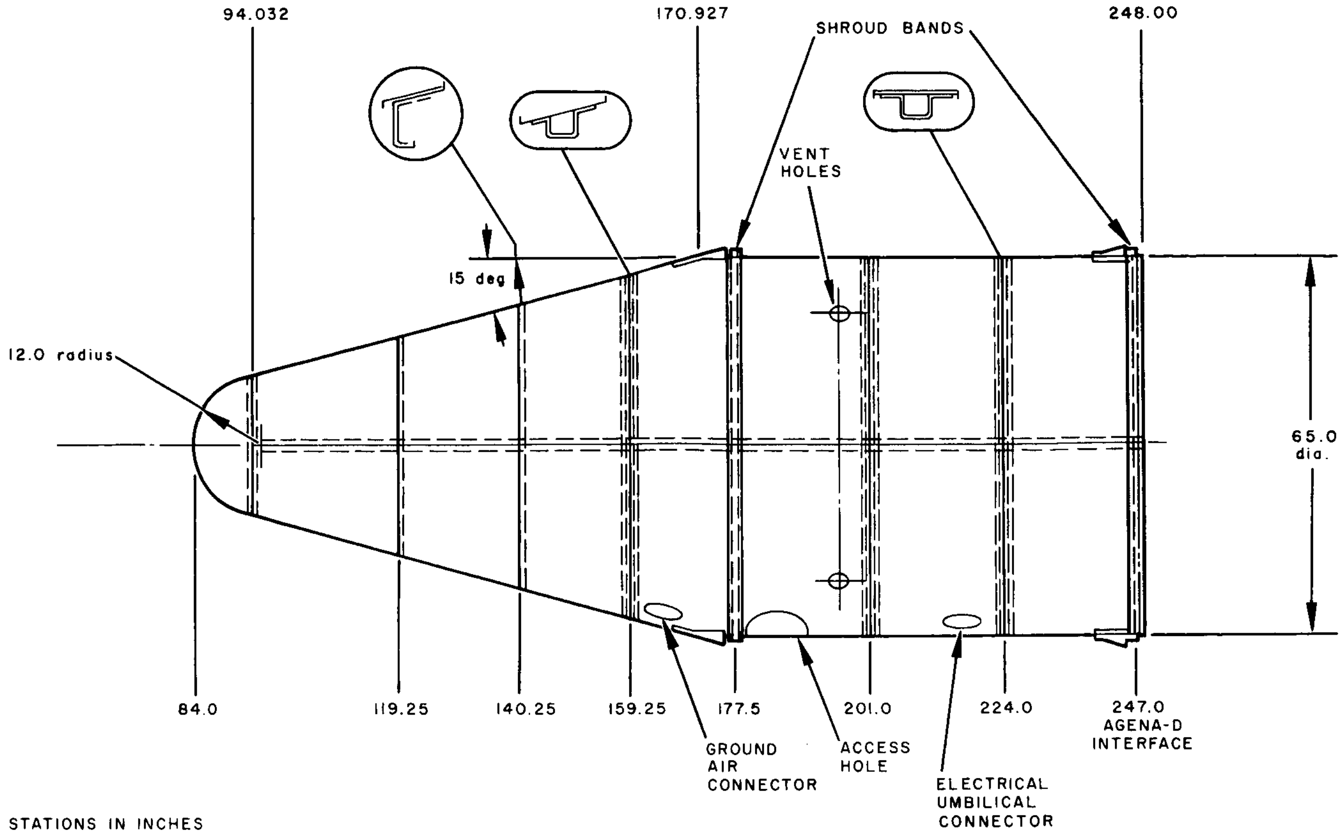


Figure 8-16. Spacecraft Shroud Configuration

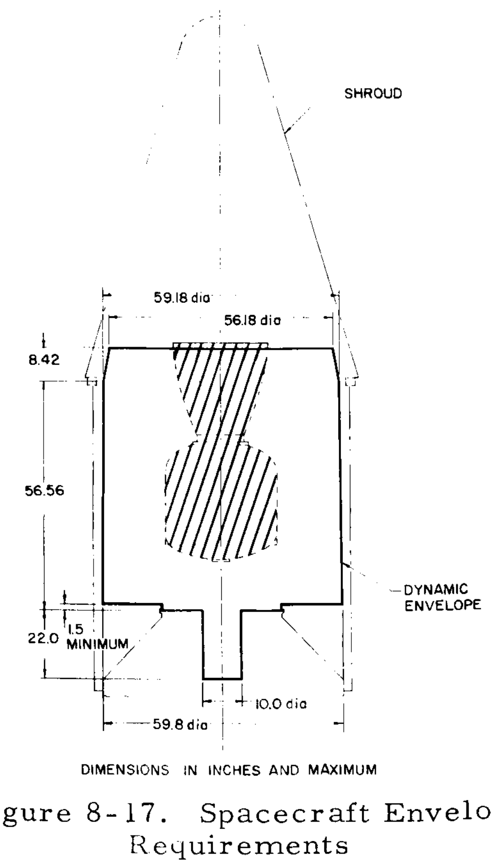


Figure 8-17. Spacecraft Envelope Requirements

plane input in the range of the predominate spacecraft lateral and longitudinal frequencies (for the sinusoidal excitation only).

3.2 Ground Testing and Handling Equipment Mechanical Interface

3.2.1 Ground Test and Handling Equipment Objectives

3.2.1.1 Spin Test Fixture: A spin test fixture for the performance of antenna tests on a spinning spacecraft was designed and is being utilized.

3.2.1.2 Mobile Assembly Fixture: A fixture to hold the basic structure during all phases of assembly is available for use. This fixture is sufficient for the study, but a more versatile fixture will be designed for the production program. This fixture will also be used to hold the spacecraft during checkout operations, and at any time maintenance is being performed.

3.2.1.3 Hoisting Sling: A combination spacecraft and apogee motor sling was designed and fabricated. A spreader bar design was chosen to ensure safe and efficient operation. The sling is capable of lifting the spacecraft from either end by four attach points, and the addition of a simple adapter converts it to an apogee motor hoisting sling. To ensure that no undue sudden shock loads due to lifting are transmitted into spacecraft structure or apogee motor casing, each of the four suspension cables are shock-mounted. The cable shock absorbing springs also function as load equalizers.

3.2.1.4 Clamp: A simple ring-gland type clamp has been designed and fabricated. For ease of operation, the clamp consists of four segments. This clamp is used for attaching the spacecraft to various handling, tooling, and test equipment.

3.2.1.5 Test Fixture: Two fixtures have been designed and are available for vibration testing of the spacecraft. One fixture duplicates the clamp portion of the Agena interface geometry. The spacecraft is attached to this fixture at the thrust tube. The second fixture duplicates the JPL motor attachments.

GROUND CONTROL EQUIPMENT

The NASA Syncom II design review of April 1963 established a basic philosophy which in broad terms describes the ground support system. The following tenets are the primary structure of this philosophy:

1) Satellite test equipment and ground station equipment will be identical to the largest practical extent.

2) Inherent in the ground support equipment will be a mechanization of data acquisition including digital printout.

3) System test equipment shall be capable of exercising all communications frequencies simultaneously.

Some of the consequences of this philosophy are immediately apparent, such as duplication of communication test equipment for the four-channel exercise; most, however, are dependent largely on good engineering judgement and a careful analysis of overall system test requirements. To facilitate this study, the system requirements were broken down into subsystems as follows:

1) TM Receivers
2) TM Transmitters
3) System Integration
4) Gas System
5) Digital Recording and Indicating
6) Analog Recording and Indication
7) Ground Controller
8) TM Decommutator
9) Command Subsystem
10) Interconnections, Wiring, and Patch
11) Communications Electronics
12) RF Subsystems
13) Communications Digital Electronics

14) Power Supplies

15) Mechanical Systems

As an aid in fulfilling the ground support equipment philosophy, a document containing system test descriptions to fulfill both specification requirements as well as qualitative demonstrations was prepared. Among the accrued benefits of the document, in addition to test descriptions, are the extrapolated list of required spacecraft test points and additional insight into test equipment configuration. Copies of the system test document and test point summary are presented later in this section.

As a result of the above document, the general ground system equipment philosophy and technical requirements, a block diagram was prepared illustrating the equipment necessary to fulfill these functions as envisioned at this time. The block diagram is not functional, since the multiple usage equipment is extensive and such a drawing would not fulfill its purpose. The block diagram states the general availability, through patching facilities, of the equipment indicated.

A byproduct of the system test list and equipment block diagram is a master index of major items comprising the subsystems indicated above. The derivation was accomplished by crosschecking the various tests and technical requirements to avoid duplications. The master index is presented later in this section.

The final portion of this summary concerns equipment peculiar to the ground stations; in particular, a cost study of van versus inflatable housing has been prepared.

<u>System Tests</u>

The following material is a preliminary study of the Advanced Syncom system test requirements, and is intended to fulfill several objectives, the most significant of which are:

1) Define test equipment requirements.

2) Suggest spacecraft access requirements.

3) Define handling and test fixture requirements.

4) Define data recording requirements.

5) Define equipment requirements for Advanced Syncom ground stations.

The test equipment master index, included with the report, is a compilation of equipment needs derived from this study. The requirements of the special equipment to be designed by Hughes have been established and preliminary block diagrams have been prepared (Figures 8-18, 8-19, and 8-20). Work is currently under way on development of some of the special circuitry involved. Work is scheduled to start on writing of detailed specifications for the support equipment components. Several make-or-buy decisions are still pending completion of studies of availability of commercial equipment.

Spacecraft test access requirements have been discussed. Table 8-14 is a tentative list of the test points that will be available for system test.

Discussions have been held concerning the requirements of spacecraft handling and special test fixtures. Table 8-15 lists tentative design objectives for the spin machine/test fixtures to be used for system test. Preliminary discussions have also been held concerning the development of a light source to be used as a solar sensor illuminator.

Preliminary studies have been made of the requirements of data recording. The use of the data to be taken has been organized as follows:

1) Quick-look

2) Detailed analysis

3) Permanent records

4) Report writing

The data to be taken in each test has been analyzed to assure that it will fulfill all of the above requirements. In some tests the data will be taken in two forms, such as "x-y" and on some storage medium such as magnetic and/or teletype tape.

The last section of this study includes a preliminary analysis of the tests required at the Syncom ground sites to assure that the ground station is operating properly.

The tests outlined in this document make use of automatic data recording wherever applicable. The automatic data recording equipment will be utilized to make tests that would otherwise be impractical. Such a test as transponder sensitivity measurements, in which the RF power output of the transponder is measured as the input RF power is varied, will produce a plot containing a hundred or so points. The intent of the system tests is to produce informative, unambiguous tests of sufficient accuracy for adequate determination of system performance.

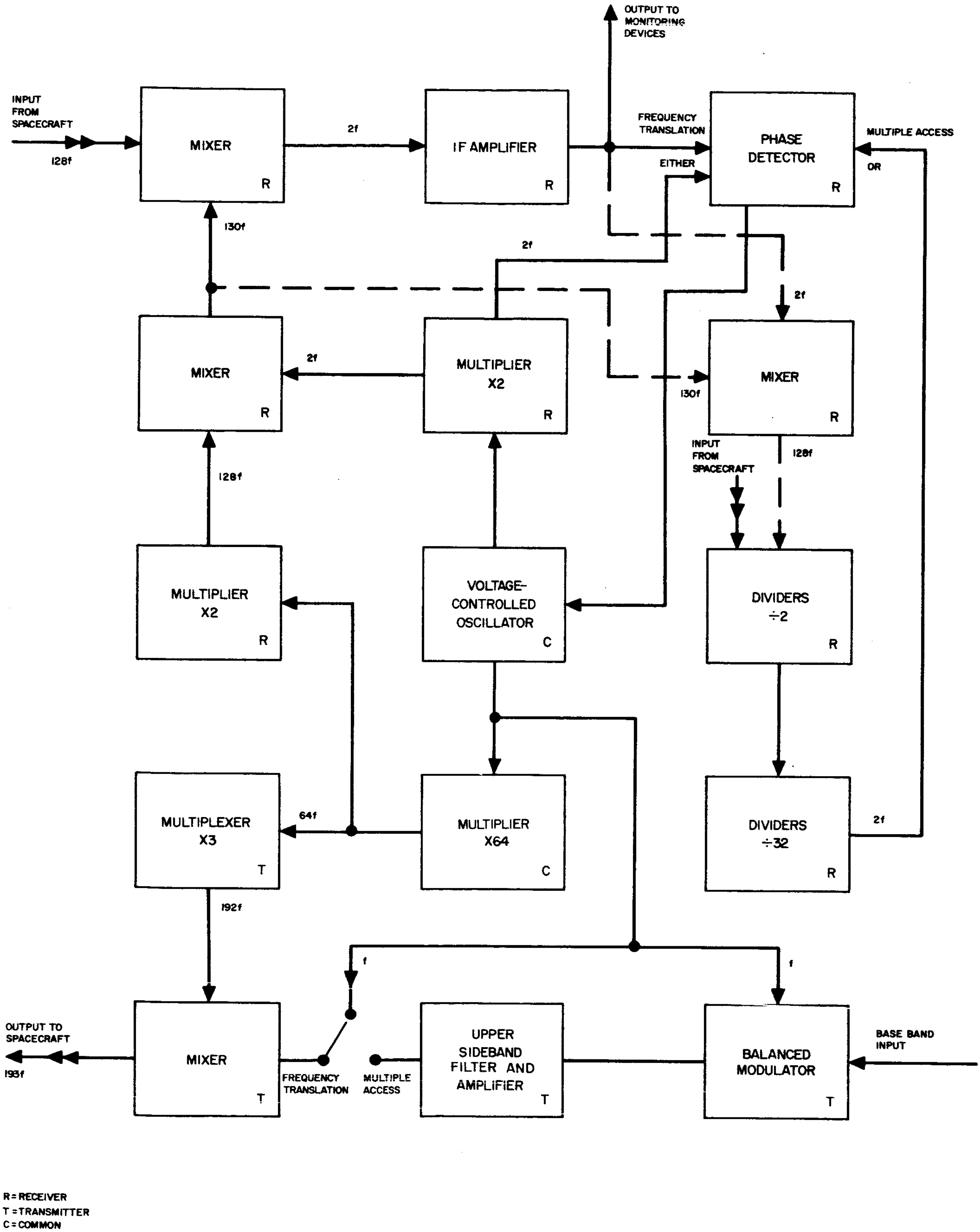


Figure 8-18. Phase-Locked Dual-Mode Receiver

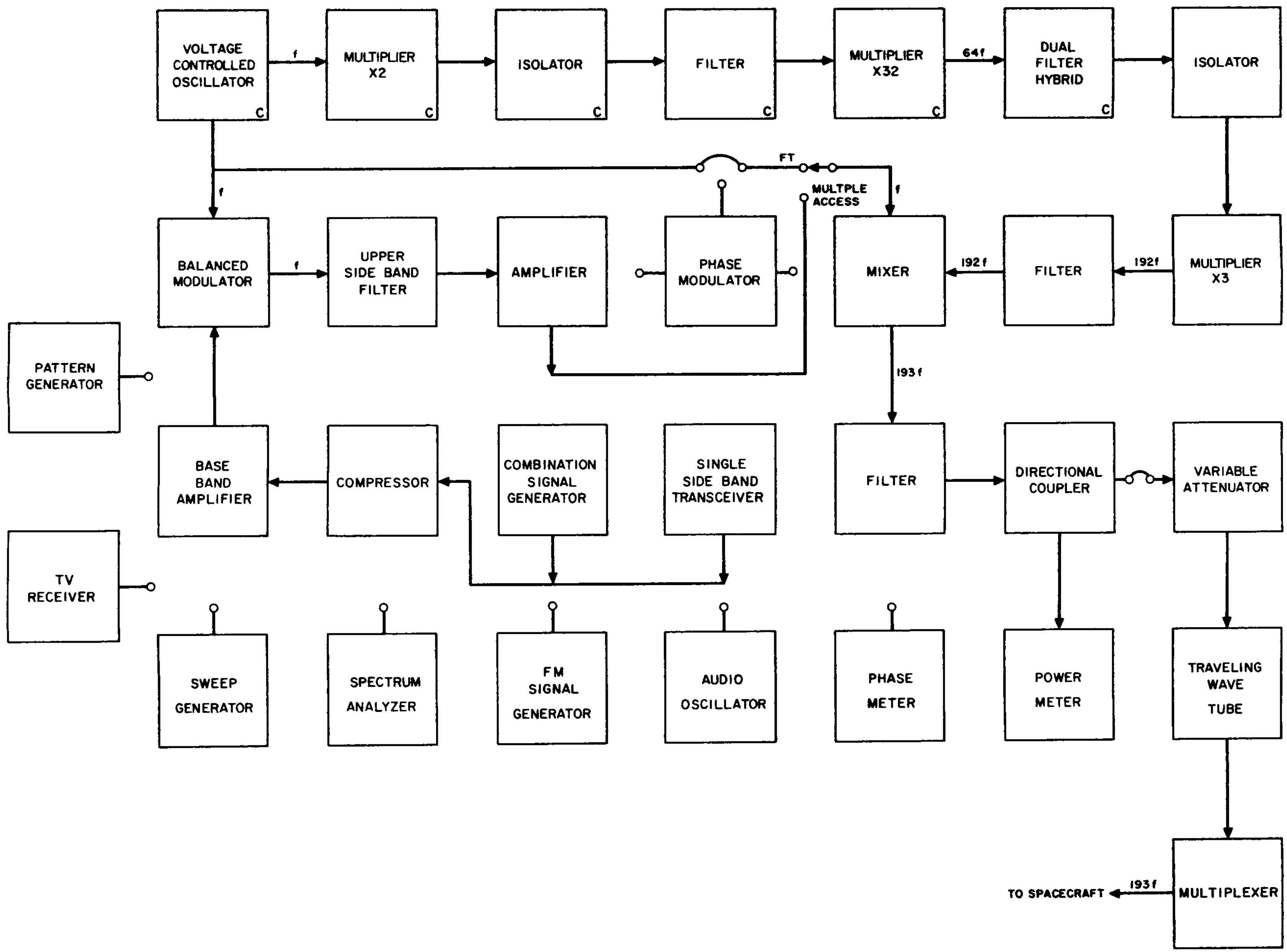
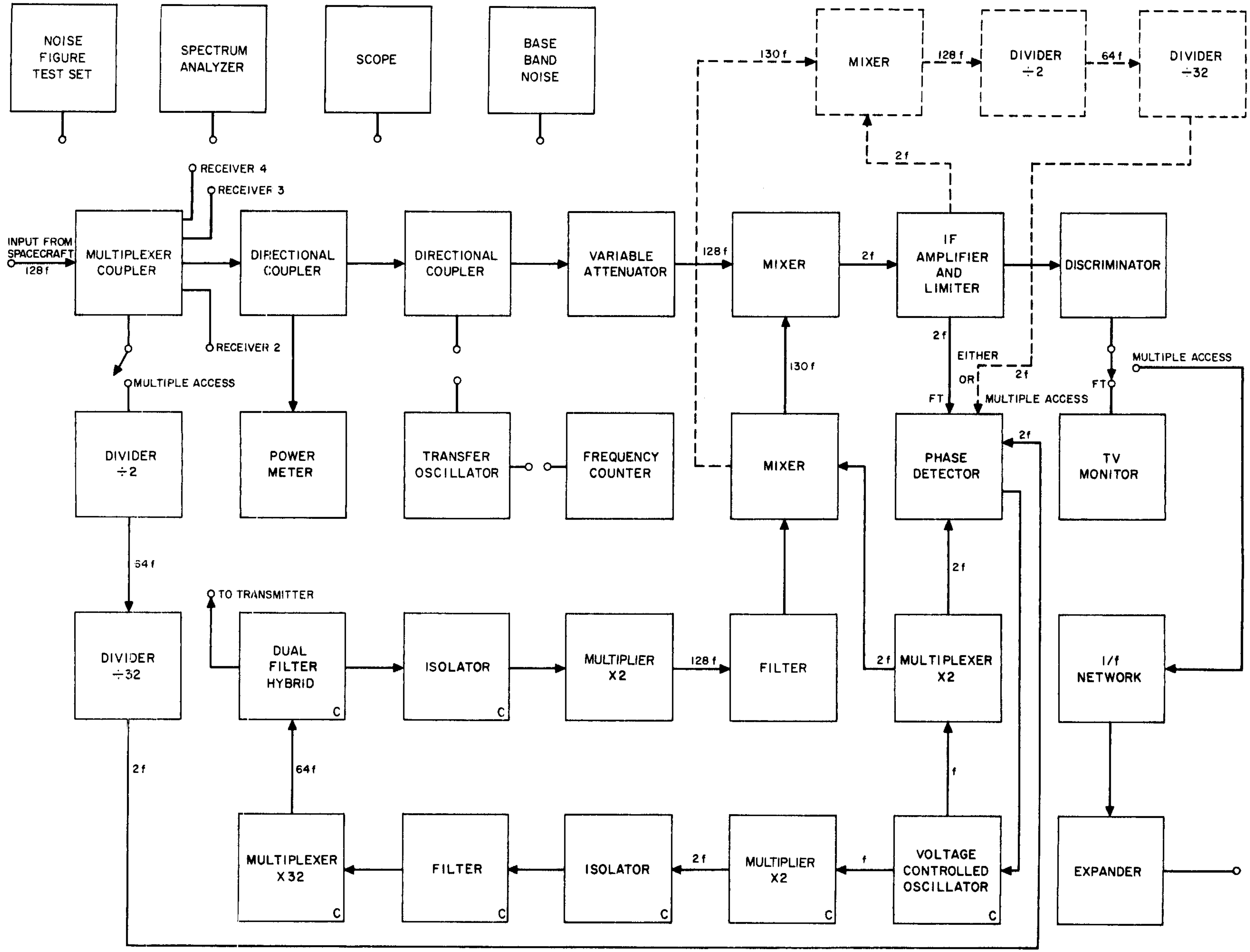


Figure 8-19. Dual-Mode Transmitter



DOTTED LINES INDICATE ALTERNATE METHOD OF MULTIPLE ACCESS ϕ LOCK
C = COMMON TO TRANSMITTER

Figure 8-20. Dual-Mode Receiver

A prime purpose of all system tests is to demonstrate that important system parameters remain effectively constant through the duration of the test program. In addition, the system tests will demonstrate that the Advanced Syncom spacecraft electronics system parameters meet all applicable specifications.

The tests outlined in this study are not intended as firm system outlines but as guides that will be constantly under consideration and evaluation as the spacecraft and support equipment evolve.

TABLE 8-14. SUGGESTED SPACECRAFT ACCESS

Communication Transponders

- RF output will be available at outputs of four directional couplers placed between ferrite switch outputs and inputs to multiplexer
- RF input will be through single directional coupler placed between antenna and multiplexer

Telemetry Subsystem

- Telemetry transmitter output/command receiver input will be available at output of directional coupler between each diplexer and balun (total of four outputs)
- Audio output of each command receiver will be available

PACE Electronics

- VCO output and F-100 output (VCO should be 512 cycles between F-100 pulses when FLL is locked)
- Input to apogee timer

Power Supply

- Unregulated bus input
- Access to charge battery

Propulsion Subsystem

- Input to each pyrotechnic squib

TABLE 8-15. SPIN MACHINE DESIGN OBJECTIVES

Mobility
• Consider self-propelled fixture
• Provide swivels - all wheels
• Provide pneumatic tires or equivalent to reduce vibration inputs to spacecraft
• Provide for fork lift and crane hoist attachments, both with and without spacecraft installed
• Provide means of braking the motion of the fixture
Support
• Provide attachment at apogee motor interface with spacecraft antenna up; should be quick-disconnect type of fastener
• Provide load capability for spacecraft less apogee motor
• Provide jacks and level for stability and leveling spacecraft mounting interface*
Rotation
• Provide for spinning spacecraft up to 150 rpm
• Spin control continuous 0 to 150 rpm*
• Spin stability 0. 15 degree per revolution or 0. 05 percent
• Spin up and braking loads not to exceed acceptance level loads on spacecraft, but spin up and braking should be as quick as possible within above limits
• Fixture should operate properly with nominally unbalanced spacecraft without excess nutation or vibration*

*Requirements such as tolerances, signal levels, impedances, etc., will be specified at a later time.

TABLE 8-15. (continued)

Power and Instrumentation

- Provide one power input to fixture
- Provide spin speed monitoring signal*
- Provide azimuth angle readout (digitized) with 0.1 degree resolution referenced to sun sensor illuminator*
- Provide electrical patch panel to accommodate spacecraft signals and fixture signals*
- Provide 24 slip rings for spacecraft connection; also consider RF slip rings*
- Limit noise levels in drive, instrumentation, and slip rings to state of the art

Accessories

- Provide scaffolding for spacecraft access
- Provide safety baffles for personnel protection while spacecraft is spinning; may be part of scaffolding
- Provide support for solar panel string illuminator*
- Provide capability of placing test equipment for RF tests in close proximity to spacecraft; scaffolding may serve this purpose
- Provide means of accurately determining position of spin axis in relation to other test devices such as antenna horns--possibly Navy azimuth circle or transit

*Requirements such as tolerances, signal levels, impedances, etc., will be specified at a later time.

TABLE 8-15. (continued)

Environment • Test stand is intended for all nonenvironmental testing but should be capable of operating in vacuum and in thermal environment • Suggestions are solicited for additional or modified design features that may be incorporated in this equipment to extend its utility for systems testing

PRELIMINARY SYSTEM TESTS

INDEX

INDEX (continued)

FREQUENCY TRANSLATION TRANSPONDERS - TEST 1

Purpose of Test

Determine the frequency and stability of frequency translation transponder master oscillator.

Source of Test

NASA Specification S2-100, paragraph 3.9.3.3.

Procedure

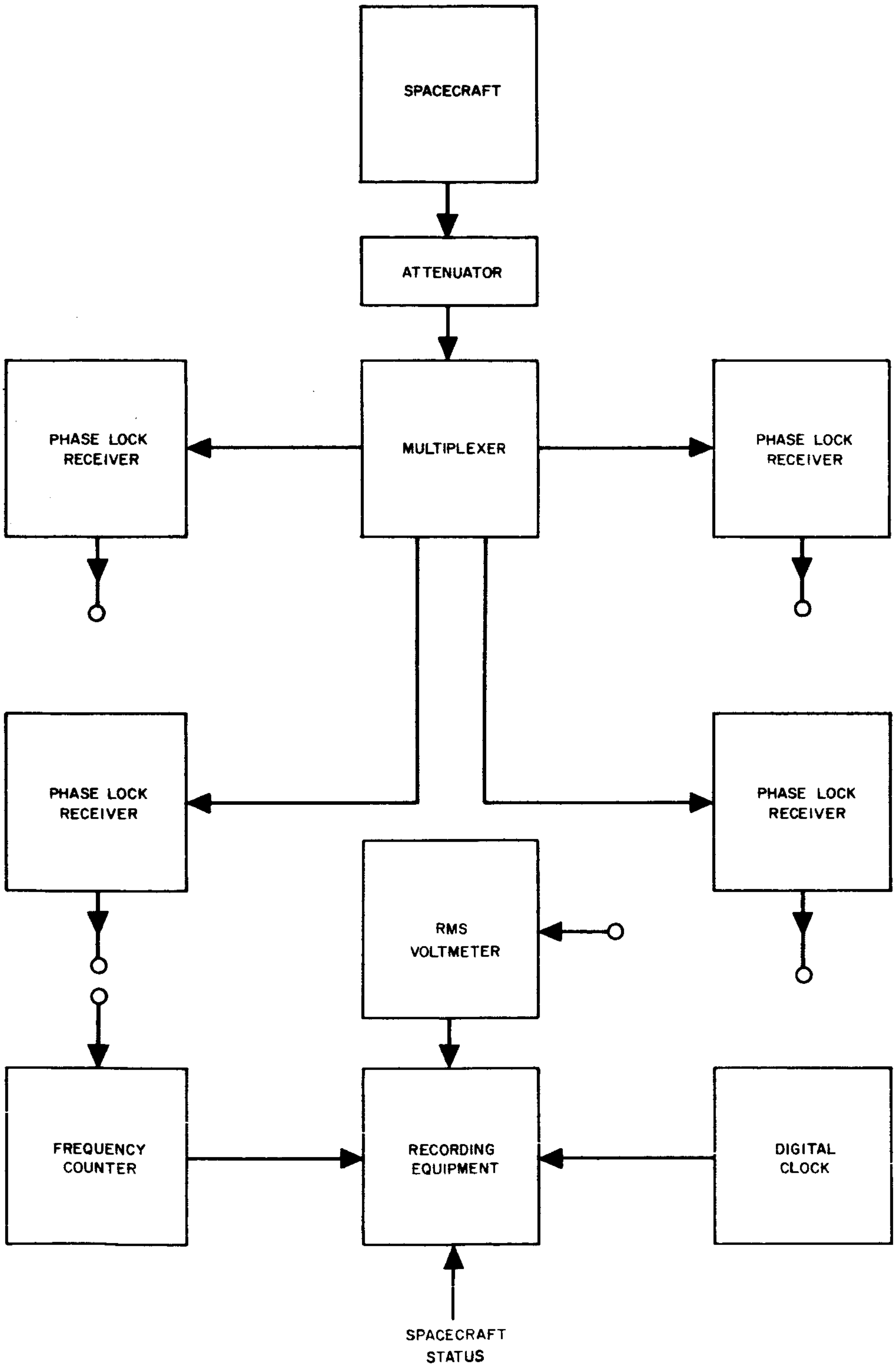
This test will be performed by measuring the frequency of the test equipment microwave transceiver local oscillator which is phase-locked to the spacecraft master oscillator. The long-term stability of the spacecraft master oscillator will be determined by recording the frequency of the phase-locked oscillator over a period of minutes. The short-term stability of the master oscillator will be determined by measuring the phase-locked oscillator control voltage with an rms voltmeter. Assuming the short-term stability of the VCO is of the same order of magnitude as spacecraft master oscillator, half the rms value of the short-term instability can be attributed to the spacecraft master oscillator. To the extent that test equipment oscillators can be fabricated which are better than spacecraft oscillators, the indicating capacity of the equipment will be improved.

Equipment Required

1) Phase-locked microwave receiver for each transponder
2) Frequency meter
3) Data recorder
4) Digital clock
5) Strip chart recorder
6) RMS voltmeter

Spacecraft Access Required

1) Transponder RF output


SPACECRAFT
ATTENUATOR
PHASE LOCK RECEIVER
MULTIPLEXER
PHASE LOCK RECEIVER
PHASE LOCK RECEIVER
PHASE LOCK RECEIVER
RMS VOLTMETER
FREQUENCY COUNTER
RECORDING EQUIPMENT
DIGITAL CLOCK
SPACECRAFT STATUS

FREQUENCY TRANSLATION TRANSPONDER - TEST 2

Purpose of Test

Measure the power output of each TWT.

Source of Test

NASA Specification S2-100, paragraph 3.9.4.2

Procedure

The power output of each TWT will be monitored by measuring the power out of a test directional coupler which is built into the spacecraft. The spacecraft test directional coupler will be located between the output of the TWT power switch and the input to the multiplexer. The power output of the redundant TWTs will be measured by turning them on sequentially.

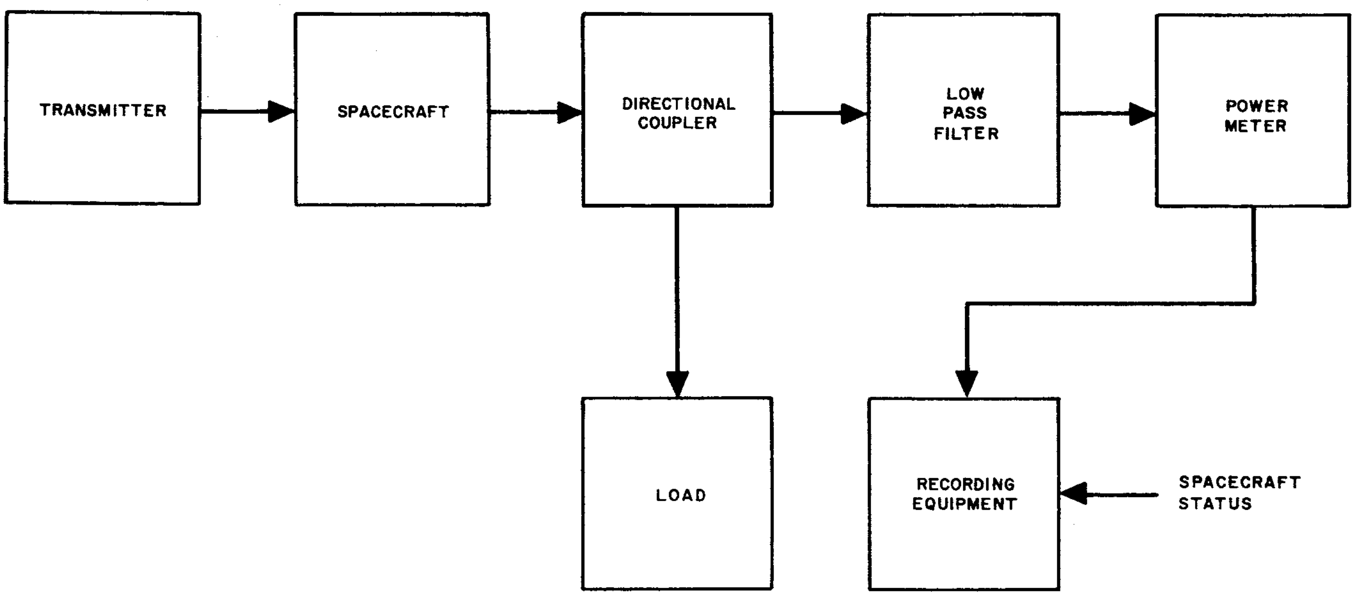
A signal will be supplied to the spacecraft receiver, with sufficient power to assure that the receiver is in limiting, and therefore assure the TWT is saturated. A low pass filter will be placed before the power meter so that power indicated will not include the harmonics.

Equipment Required

1) Power meter
2) Directional coupler and loads
3) Data recorder
4) Low pass filter

Spacecraft Access Required

1) Transponder RF output


TRANSMITTER
SPACECRAFT
DIRECTIONAL COUPLER
LOW PASS FILTER
POWER METER
LOAD
RECORDING EQUIPMENT
SPACECRAFT STATUS

FREQUENCY TRANSLATION TRANSPONDER - TEST 3

Purpose of Test

Determine the sensitivity of each frequency translation transponder receiver.

Source of Test

Hughes, to ensure spacecraft performance parameters.

Test Procedure

A plot will be made of the power output of the transponder as the input signal strength is varied. This plot will be made by plotting the input signal to the transponder on the x-axis of an x-y recorder while recording the transponder output power on the y-axis. The input signal strength will be varied from -100 dbm to -50 dbm.

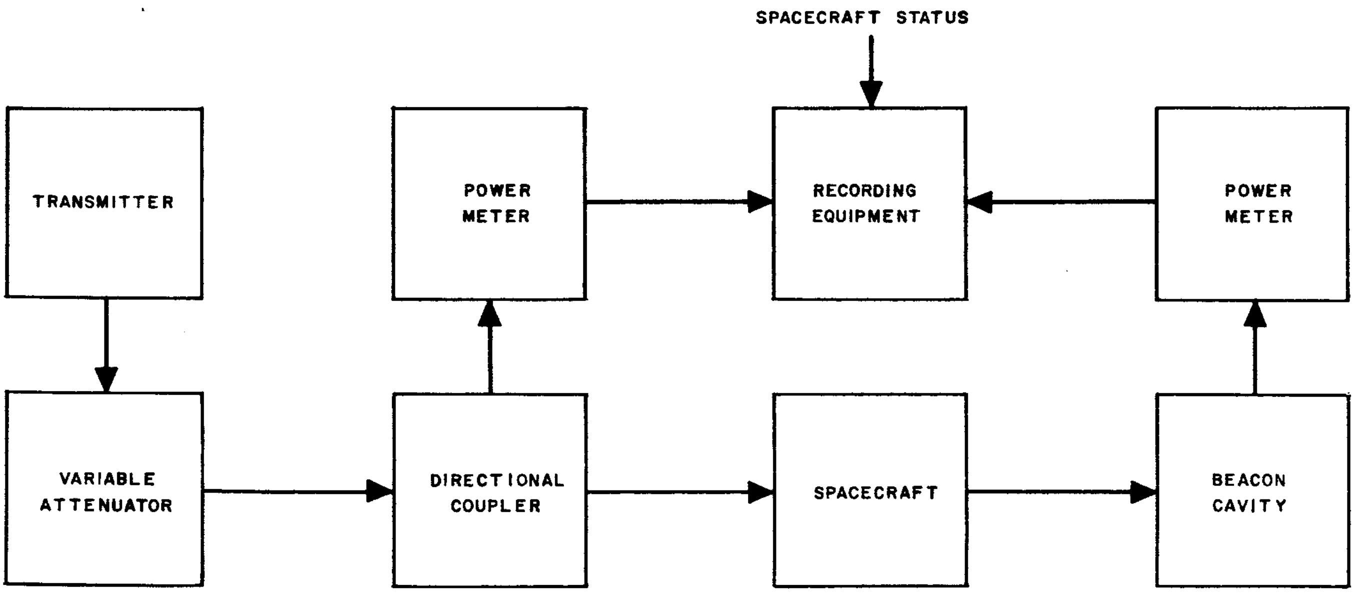
The beacon signal will be "trapped" in a cavity; therefore, power output plotted will be an indication of signal power only.

Equipment Required

1) Microwave signal generator
2) Variable attenuator
3) Data recorder
4) Tape punch
5) Power meter (2)

Spacecraft Access Required

1) Transponder RF input
2) Transponder RF output


SPACECRAFT STATUS
TRANSMITTER
POWER METER
RECORDING EQUIPMENT
POWER METER
VARIABLE ATTENUATOR
DIRECTIONAL COUPLER
SPACECRAFT
BEACON CAVITY

FREQUENCY TRANSLATION TRANSPONDER - TEST 4

Purpose of Test

Determine the noise figure of each of the frequency translation transponders.

Source of Test

NASA Specification S2-0100, paragraph 3.9.3.1.3.

Test Procedure

This test will be performed using the single frequency noise figure method. The test is conducted by measuring the noise transmitted from the spacecraft with no signal input. A signal is then applied and its amplitude is adjusted such that the total signal transmitted from the spacecraft has now doubled. The noise figure can then be computed as shown below:

$$NF = \frac{(\text{Signal/Noise})\ \text{ideal circuit}}{(\text{Signal/Noise})\ \text{actual circuit}}$$

When signal power is adjusted to equal noise power in the actual circuit

$$NF = (\text{Signal/Noise})\ \text{ideal} = P_{signal}/P_{noise}$$

$$NF = \frac{P_{signal}}{KTB}$$

Assume 25 mc bandwidth

$$NF\ (db) - P_s\ (dbm) - 198.6 + 24.8 + 74$$

$$NF\ (db) = P_s\ (dbm) - 99.8$$

Assume NF = 10 db, then

$$P_s\ (dbm) = -89.8\ dbm$$

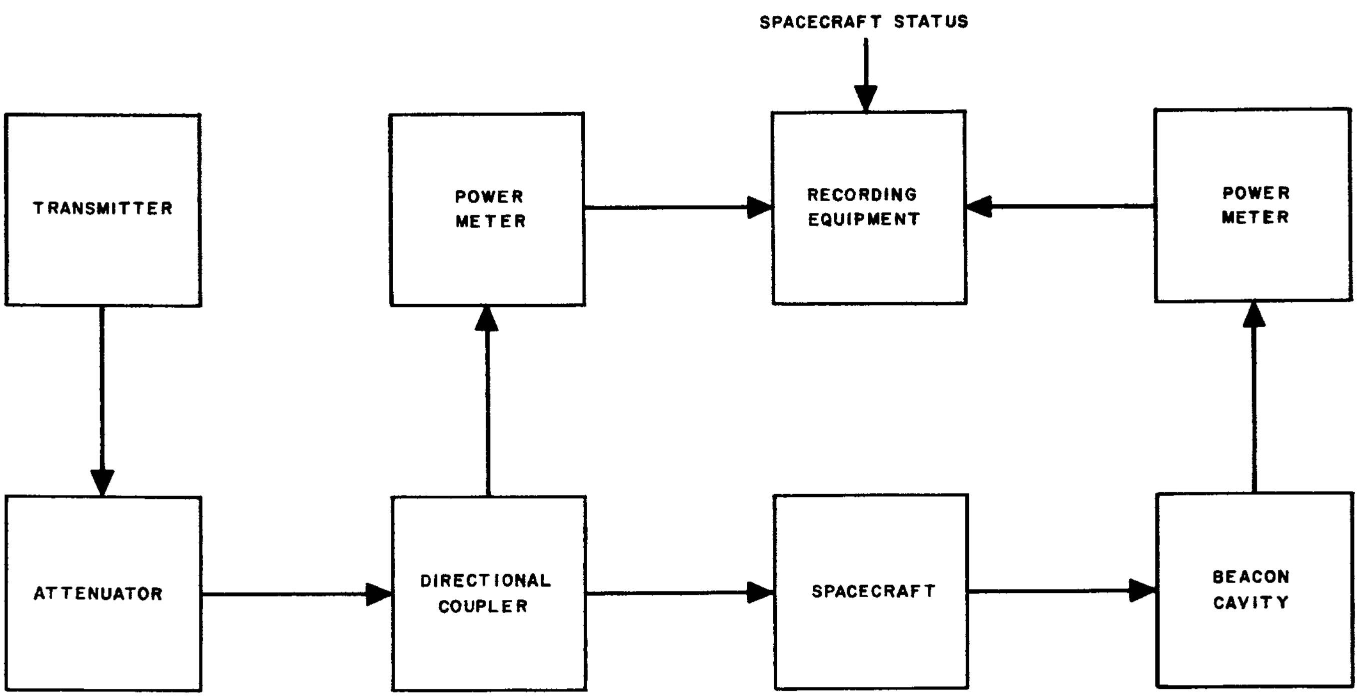
This low level of signal required should assure that the test can be made with the transponder operating in its linear range. The results of Test 3 can be used to establish that the signal level required to perform this test is in fact in the linear gain range of the transponder. The accuracy of the test would be limited by the accuracy with which P_s could be measured and would probably be ± 1 db.

Equipment Required

1) Two power meters
2) Microwave signal generator
3) 40-db directional coupler (6 kmc)
4) Beacon cavity filter
5) Precision attenuator 0 - 40 db

Spacecraft Access Required

1) Antenna coupler, receiver



FREQUENCY TRANSLATION TRANSPONDER - TEST 5

Purpose of Test

Determine the envelope delay of the frequency translation transponder.

Source of Test

Hughes, to measure spacecraft performance.

Test Procedure

The envelope delay of a circuit is defined as the derivative of its phase-frequency curve. Therefore:

$$T = \frac{d\phi}{d\omega}$$

If two measurements of phase are made at two frequencies very near each other, the derivative may be closely approximated by:

$$T = \frac{\Delta\phi}{\Delta\omega}$$

where $\Delta\phi$ represents the difference in the phases at the two frequencies $\Delta\omega$. T will be seconds when $\Delta\phi$ is in radians.

The envelope delay may be measured by supplying the transponder input two frequencies, f_1 and f_2, separated by a fixed frequency ΔF. The difference in phase shift experienced by the signals f_1 and f_2 is a measure of $\Delta\phi$.

The figure is a simplified block diagram of how such a test may be implemented. Instead of measuring the absolute phase shifts of f_1 and f_2, the phase shift of the "beat note" between f_1 and f_2 is determined.

Therefore, $\Delta\phi$ is the phase difference in degrees between the beat note before and after translation through the spacecraft.

Therefore,

$$T = \frac{\phi}{360(\Delta F)}$$

If it is assumed that

$$T = 10^{-9} \text{ second}$$

and the phase meter can resolve phase errors on the order of 0.05 degree, then the smallest usable ϕ = 0.1 degree, and

$$F = \frac{\phi}{(360)(T)} = \frac{0.1}{(360)(x10^{-9})}$$

$$= 278 \text{ kc}$$

Therefore, F is small compared to the total 25 mc bandwidth.

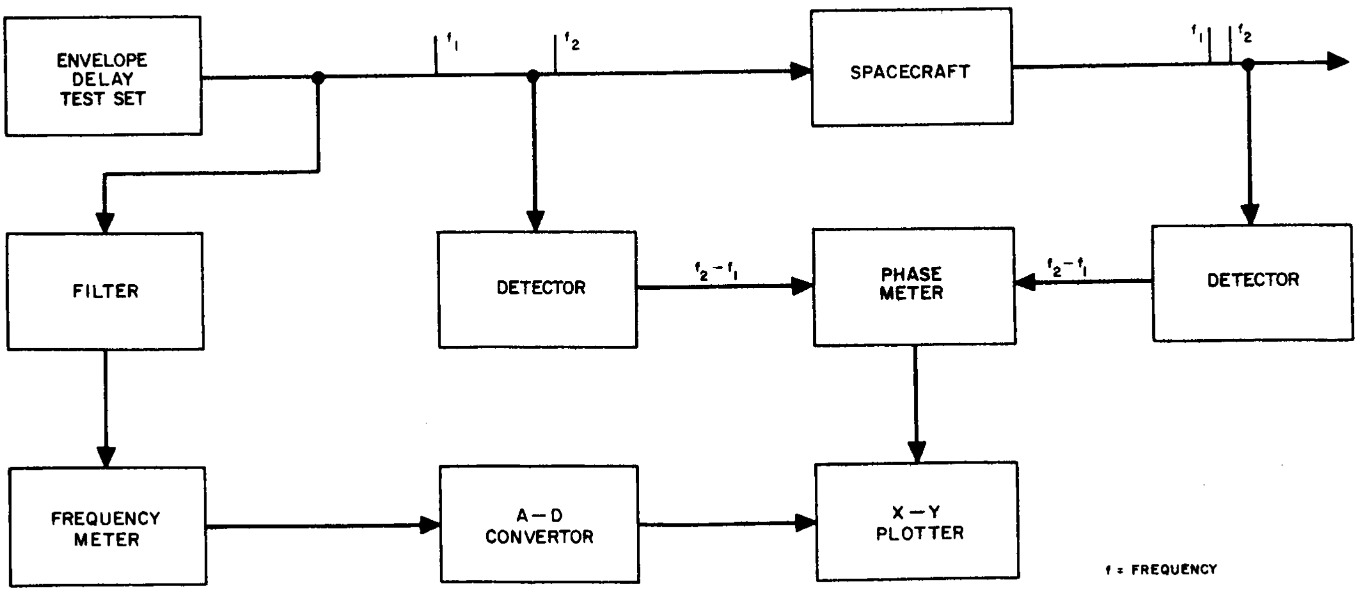
The x-y plotter will be used to plot ϕ on the y axis and $\frac{f_2 + f_1}{2}$ on the x axis.

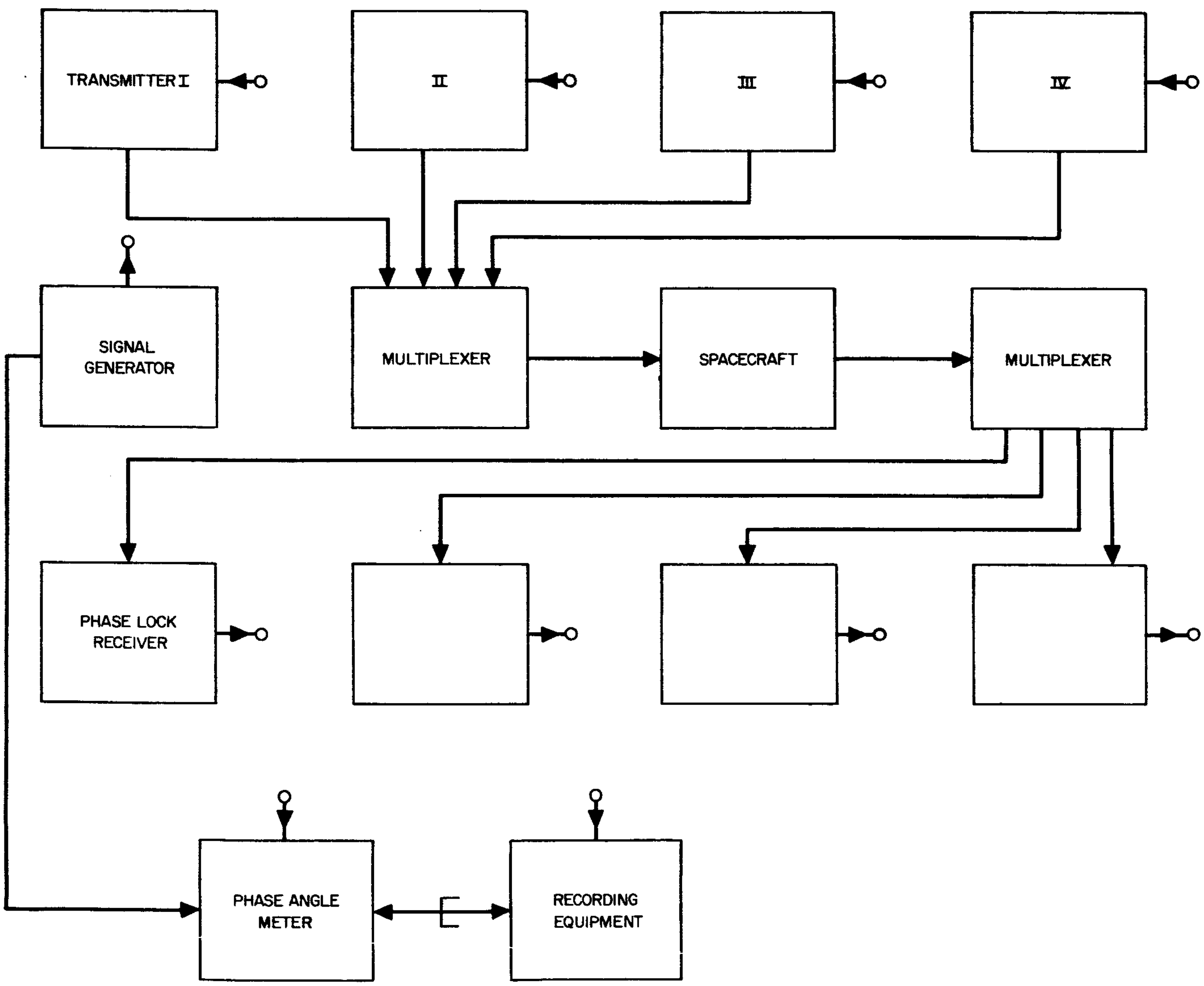
Equipment Required

1) Envelope delay test set
2) Phase meter (if not part of envelope delay test set)
3) Recording equipment
4) Frequency counter

Spacecraft Access Required

1) RF input
2) RF output




TRANSMITTER I
II
III
IV
SIGNAL GENERATOR
MULTIPLEXER
SPACECRAFT
MULTIPLEXER
PHASE LOCK RECEIVER
PHASE ANGLE METER
RECORDING EQUIPMENT

FREQUENCY TRANSLATION TRANSPONDER - TEST 6

Purpose of Test

Ensure that there are no undesirable spurious signals or intermodulation products being transmitted from the spacecraft.

Source of Test

Hughes, to ensure spacecraft performance.

Test Procedure

With all transponders on in the frequency translation mode but with no signals applied to the spacecraft, the output spectrum of each transponder will be examined for spurious signals, and the noise spectrum will be examined for proper shape. A photograph record will be made of each transponder spectrum.

A signal will then be applied to all transponders simultaneously and the output spectrum of each transponder will again be photographed. This test will be made under standard conditions; i.e., the same signals will be applied to the spacecraft each time the test is performed. It is felt that the spectrum analyzer does not adequately portray the intermodulation products transmitted by the spacecraft since the analyzer generates intermodulation products in its own mixer. Therefore, with the test made under standard conditions, the results (the spectrum photographs) of the test will be compared with the results of a control test.

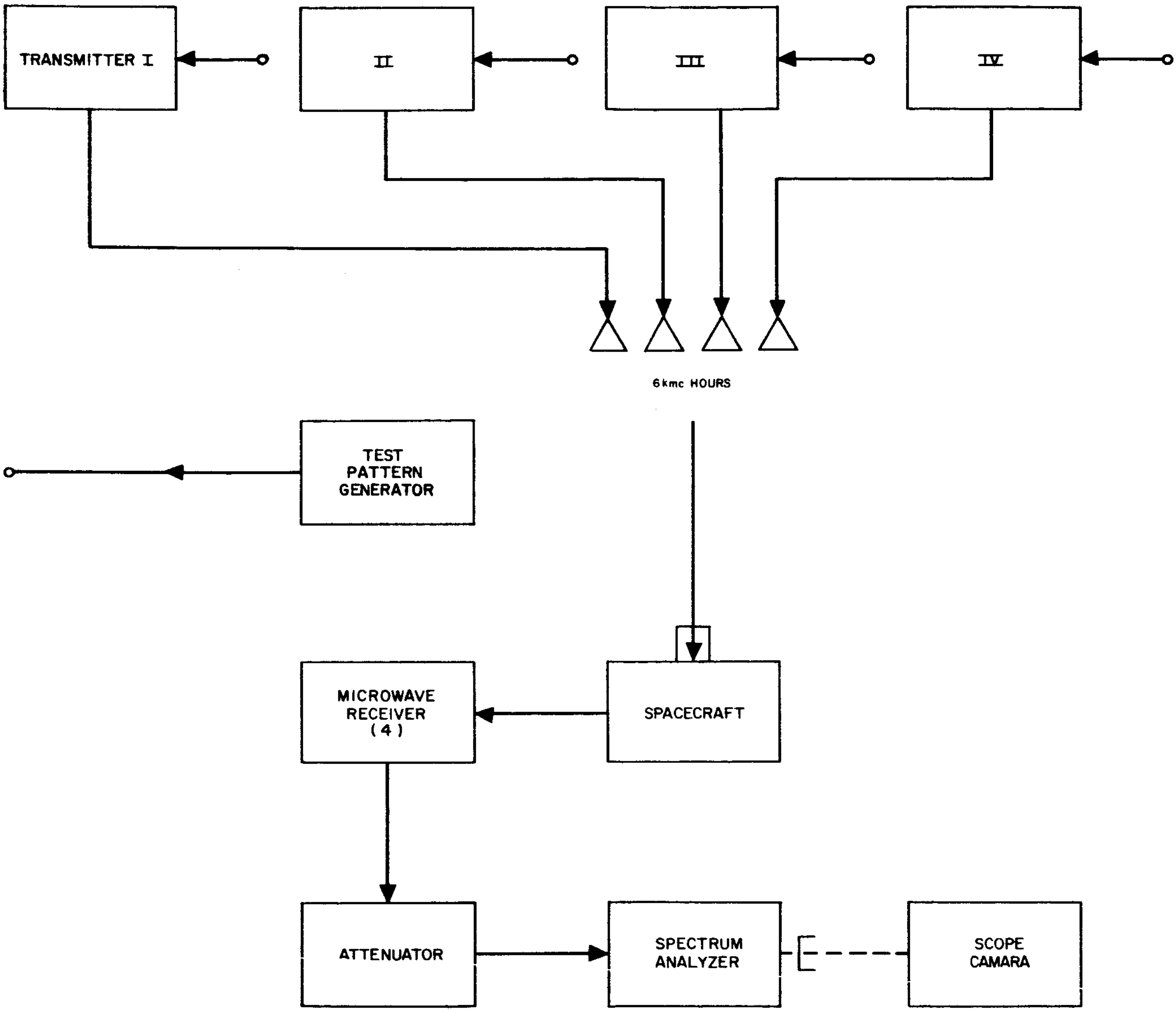
The signal to one transponder will then be removed and the IF spectrum of its associated receiver will be examined for crosstalk products. The test will be repeated for each of the other transponders by turning off their input signals, one at a time.

Equipment Required

1) Four microwave signal generators
2) Spectrum analyzer
3) TV test pattern generator
4) Scope camera
5) Variable attenuator
6) Four microwave receivers

Spacecraft Access Required

1) RF input
2) RF output



TRANSMITTER I
II
III
IV
6kmc HOURS
TEST PATTERN GENERATOR
MICROWAVE RECEIVER (4)
SPACECRAFT
ATTENUATOR
SPECTRUM ANALYZER
SCOPE CAMARA

FREQUENCY TRANSLATION TRANSPONDER - TEST 7

Purpose of Test

To demonstrate the response of the frequency translation transponders to various TV test signals.

Source of Test

Hughes, demonstrate system performance.

Test Procedure

Several types of standard TV signals including a multiburst, stair-step $\sin^2$ pulse and window, and color TV test pattern will be passed through each transponder. The returned signals will be displayed on either a scope or TV monitor and photographed. These tests demonstrate the transponder response to established test signals and permit comparison to standard performance specification for wide-band repeaters.

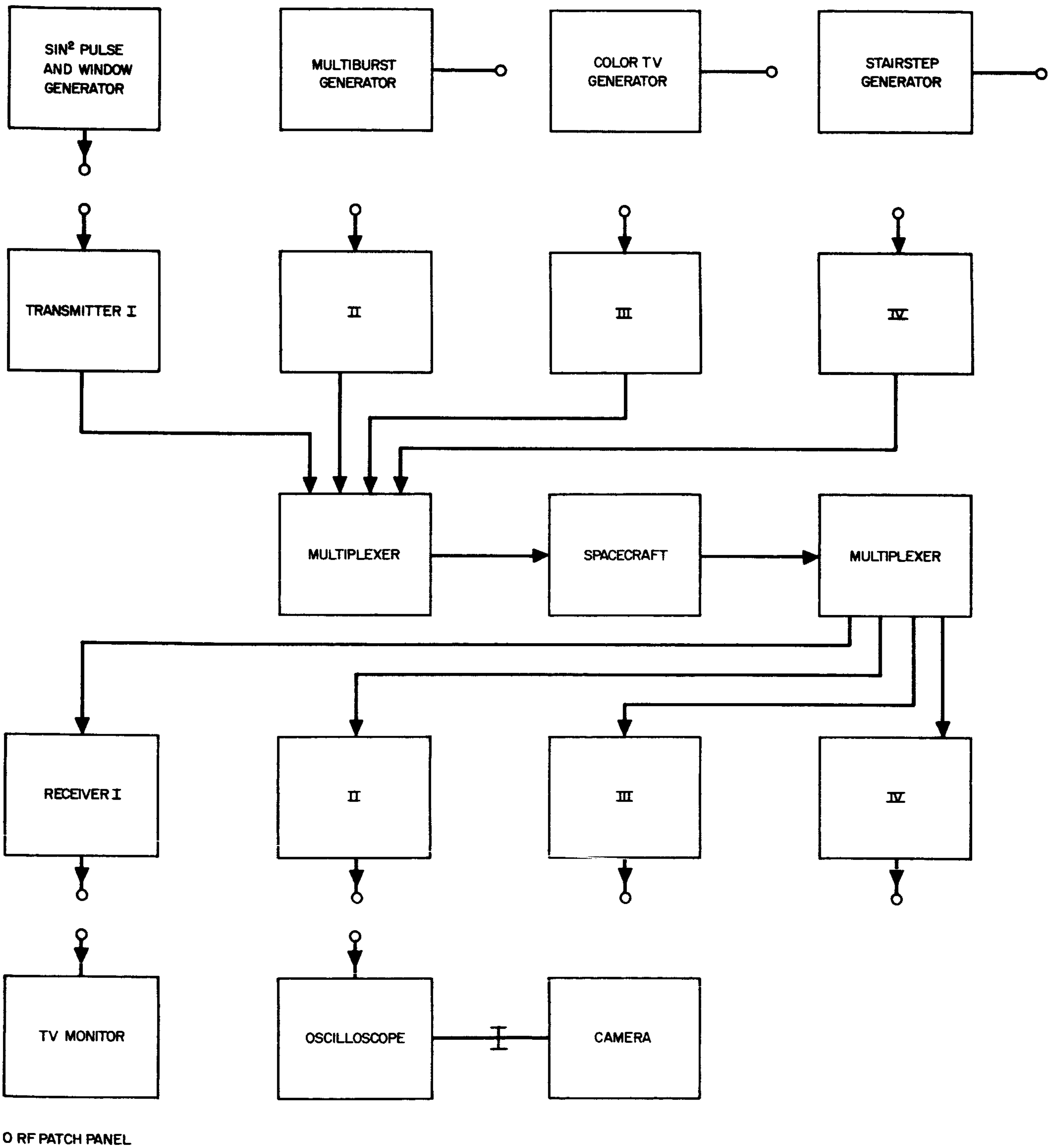
The stairstep generator comprises a low-frequency step-function video signal that will expose the amplitude-linearity characteristics of the circuit. The multiburst generator produces a low-frequency reference level pulse (white flag) followed on a time basis by a series of six bursts of known amplitude and frequency, expressly designed to demonstrate the amplitude-frequency characteristics of the circuit. The $\sin^2$ pulse and window generator output is a low-frequency square wave followed (or preceded) by a high-frequency $\sin^2$ pulse. This video signal will expose the amplitude-frequency and phase frequency characteristics of the circuit and permit waveform distortions such as ringing, overshoots, undershoots, tilts, etc., to be measured. In addition, a color TV test pattern will be applied to the system in a subjective transmission test.

Equipment Required

1) Color TV signal generator
2) Color TV monitor
3) Monitor camera
4) Multiburst generator
5) Stairstep generator
6) Sin^2 pulse and window
7) Oscilloscope
8) Scope camera

Spacecraft Access Required

1) RF input
2) RF output



FREQUENCY TRANSLATION TRANSPONDER - TEST 8

Purpose of Test

Determine the beacon power output with and without signals applied to the transponder.

Source of Test

NASA Specification S2-0100, paragraph 5.9.1.

Test Procedure

The beacon power will be measured at the output of a bandpass filter which passes the beacon signal but blocks the translated spectrum.

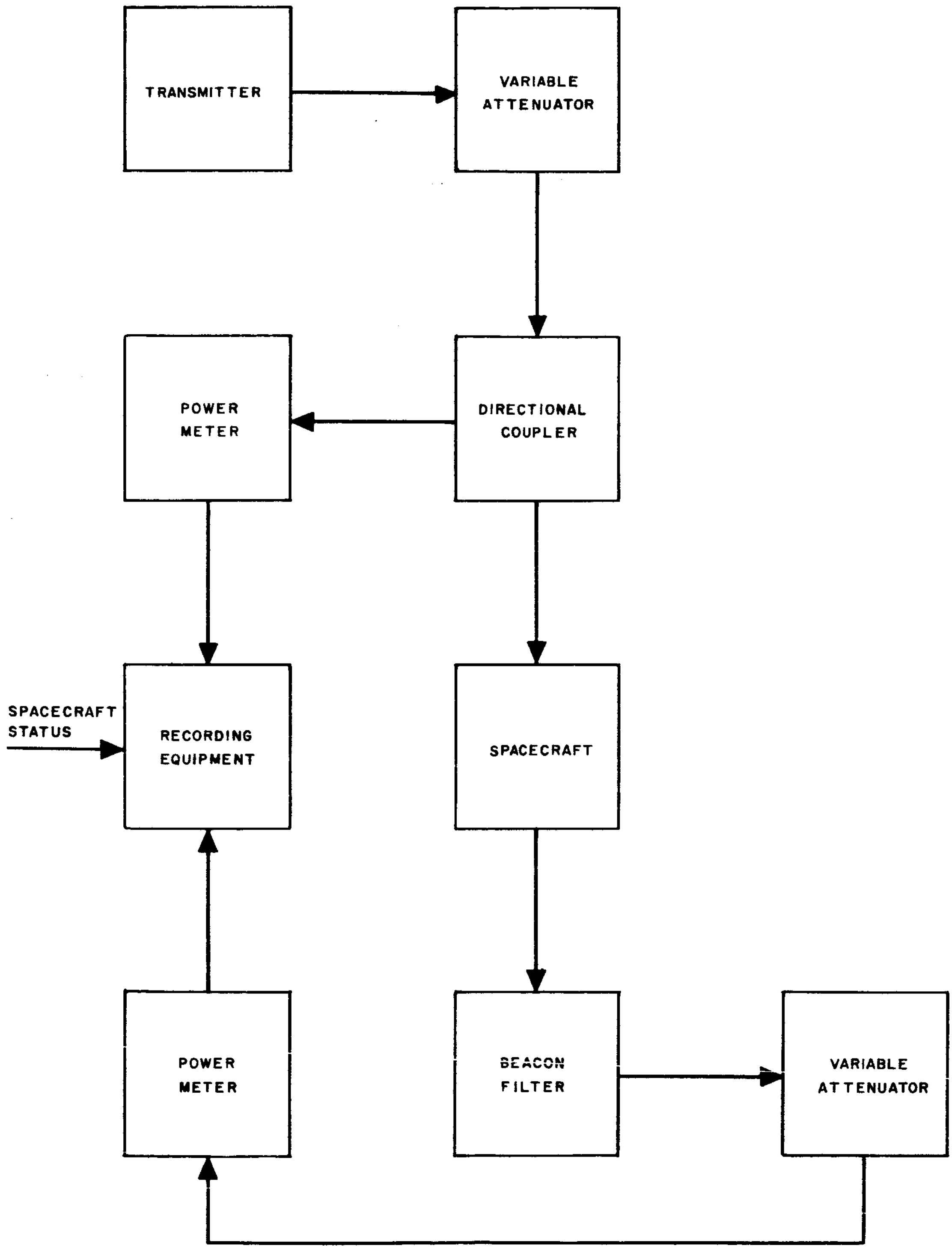
The beacon power will be measured as the input signal level to the spacecraft is varied from -100 dbm to -50 dbm. The results will be plotted on an x-y plotter.

Equipment Required

1) Power meter
2) Spectrum analyzer
3) Microwave signal generator
4) Beacon filter

Spacecraft Access Required

1) RF output
2) RF input


TRANSMITTER
VARIABLE ATTENUATOR
POWER METER
DIRECTIONAL COUPLER
SPACECRAFT STATUS
RECORDING EQUIPMENT
SPACECRAFT
POWER METER
BEACON FILTER
VARIABLE ATTENUATOR

MULTIPLE-ACCESS TRANSPONDER - TEST 9

Purpose of Test

Measure the frequency and stability of the multiple-access transponder master oscillator.

Source of Test

NASA Specification S2-0100, paragraph 3.9.3.3.

Test Procedure

This test will be performed by measuring the frequency of the test equipment dual-mode receiver which is phase-locked to the unmodulated carrier transmitted from the multiple-access transponder.

The long-term stability of the master oscillator will be determined by measuring the VCO frequency over a period of several minutes.

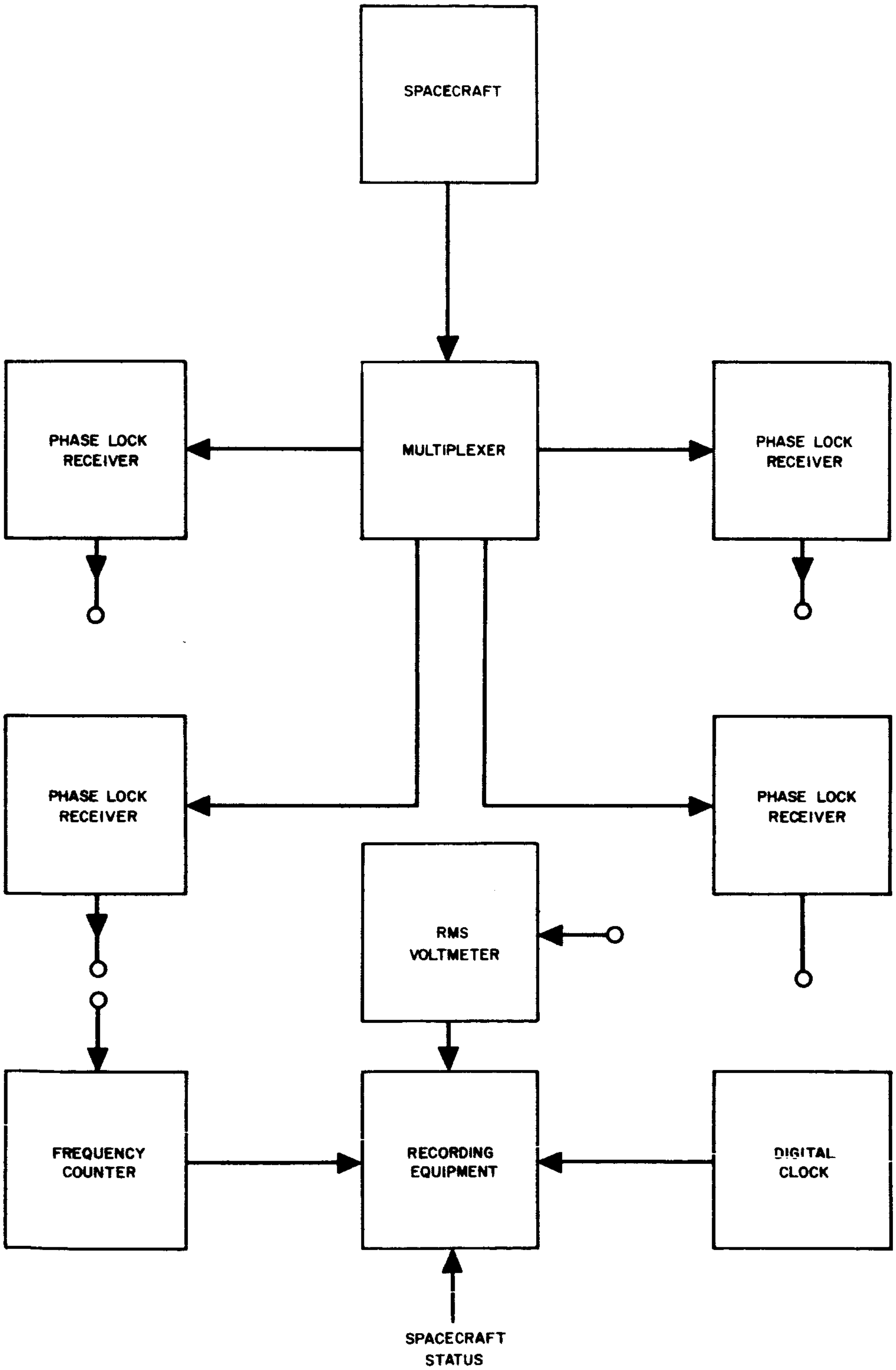
The short-term stability will be determined by measuring the VCO control voltage with an rms voltmeter. Assuming the VCO short-term stability is of the same magnitude as the master oscillator, half the rms value can be attributed to the master oscillator.

Test Equipment Required

1) Phase-locked microwave receiver
2) Frequency meter
3) Recording equipment
4) Digital clock

Spacecraft Access Required

1) RF output


SPACECRAFT
MULTIPLEXER
PHASE LOCK RECEIVER
PHASE LOCK RECEIVER
PHASE LOCK RECEIVER
PHASE LOCK RECEIVER
RMS VOLTMETER
FREQUENCY COUNTER
RECORDING EQUIPMENT
DIGITAL CLOCK
SPACECRAFT STATUS

MULTIPLE-ACCESS TRANSPONDER - TEST 10

Purpose of Test

Determine the power output of each TWT.

Source of Test

NASA Specification S2-0100, paragraph 3.9.4.2.

Test Procedure

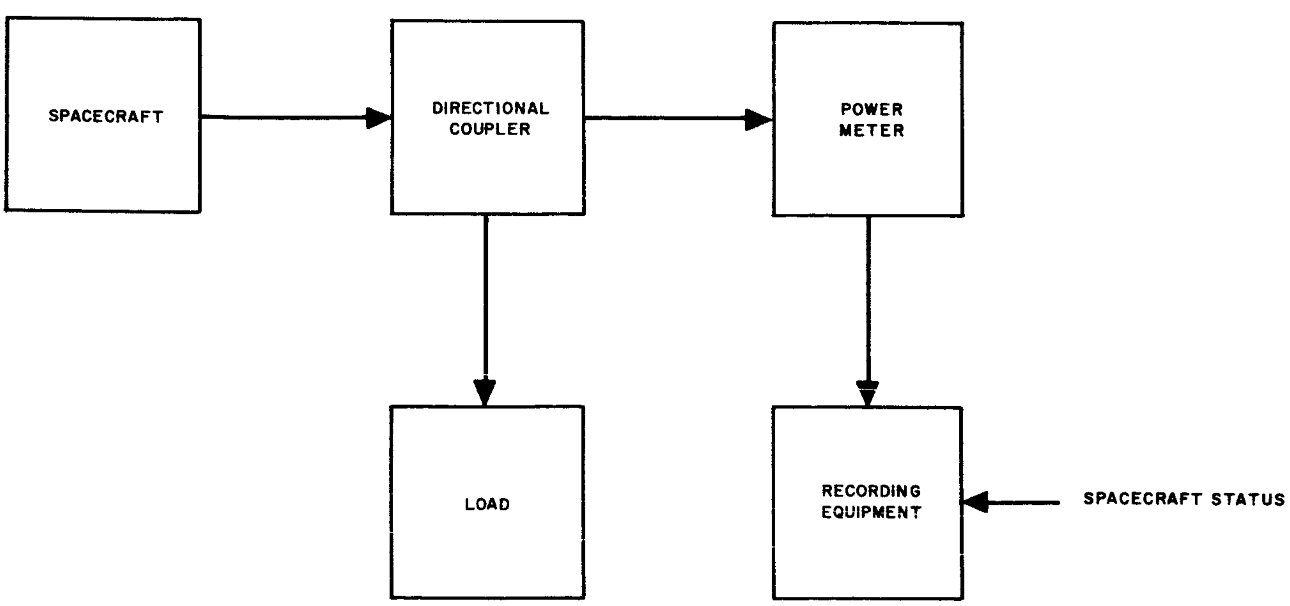
The power output of the multiple-access transponder will be measured in the same manner as that of the frequency translation transponder. The purpose in repeating the measurement here is to assure that there is sufficient TWT drive from the multiple-access transponders.

Equipment Required

1) Power meter
2) Directional coupler

Spacecraft Access Required

1) RF output



MULTIPLE-ACCESS TRANSPONDER - TEST 11

Purpose of Test

Determine the noise figure of the multiple-access transponder.

Source of Test

NASA Specification S2-0100, paragraph 3.9.3.2.3.

Test Procedure

The noise figure of the multiple-access transponder will be determined in a manner analogous to the method used to measure the noise figure of the frequency translation transponder. The first step will be to measure the amount of noise modulation on the multiple-access carrier with no signal applied to the transponder. This will be done by measuring the signal output from the dual-mode receiver discriminator. Then a single-sideband noise signal will be applied to the transponder. The signal power required to double the discriminator output power will be measured. Then noise figure will be:

$$NF = \frac{P_s}{KTB}$$

Assume NF = 10db bandwidth = 5 mc

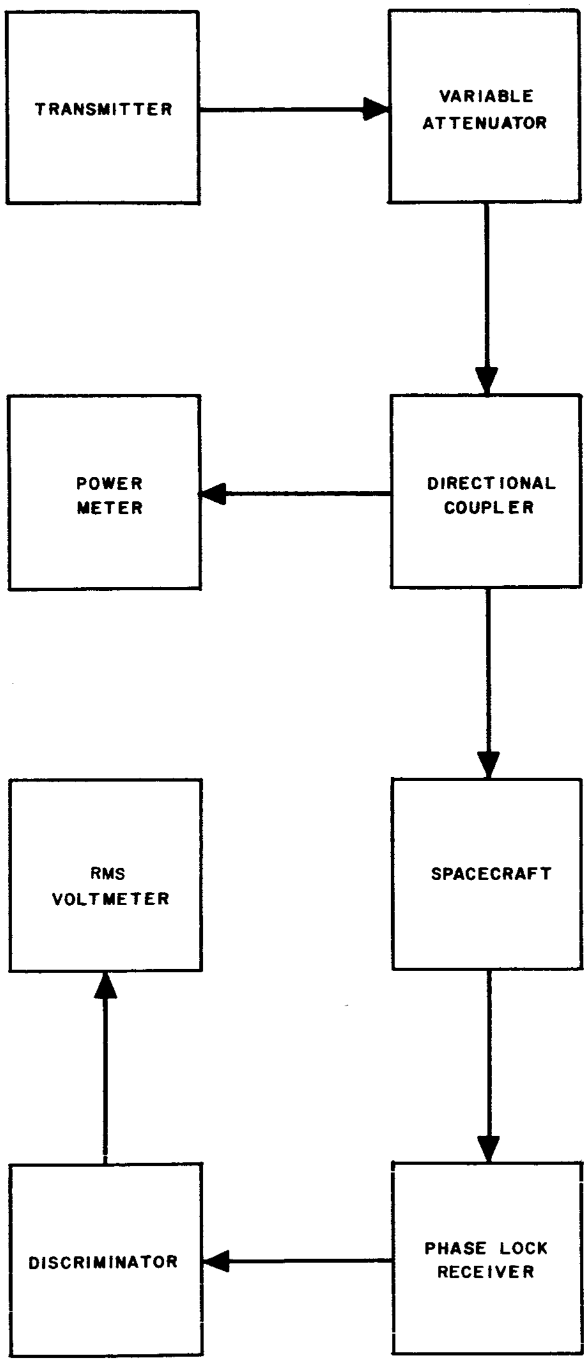
$$P_s = 97 \text{ dbm}$$

Equipment Required

1) Dual-mode transmitter
2) Power meter
3) Dual-mode receiver
4) RMS voltmeter

Spacecraft Access

1) RF input
2) RF output



TRANSMITTER
VARIABLE ATTENUATOR
POWER METER
DIRECTIONAL COUPLER
RMS VOLTMETER
SPACECRAFT
DISCRIMINATOR
PHASE LOCK RECEIVER

MULTIPLE-ACCESS TRANSPONDER - TEST 12

Purpose of Test

Determine modulation characteristics of the multiple-access transponder as function of:

a) Input signal strength
b) Number of test tones
c) Test tone frequency

Source of Test

Hughes, measure spacecraft operating characteristics.

Test Procedure

The modulation characteristics of the multiple-access carrier, as the input signal strength is varied, will be determined by measuring the output of a calibrated discriminator at the output of the test equipment dual mode receiver. The results of this test will be recorded on an x-y recorder. The function of this test is to demonstrate that the modulation index of the multiple-access transponder is a repeatable function of the input signal level.

The input signal will then be adjusted such that a signal of normally expected power is applied to the spacecraft. The amplitude of the signal at the output of the calibrated discriminator will be measured. A number of test tones (approximately 50 tones) will then be added and the amplitude of test signal will again be measured. The test is designed to demonstrate that the modulation index of a channel is independent of channel loading.

A third test of modulation index will be made by sweeping a test signal across the base band and noting its output amplitude. This test will demonstrate that the modulation index is independent of channel position in base band. The results of this test will be plotted as signal amplitude versus frequency.

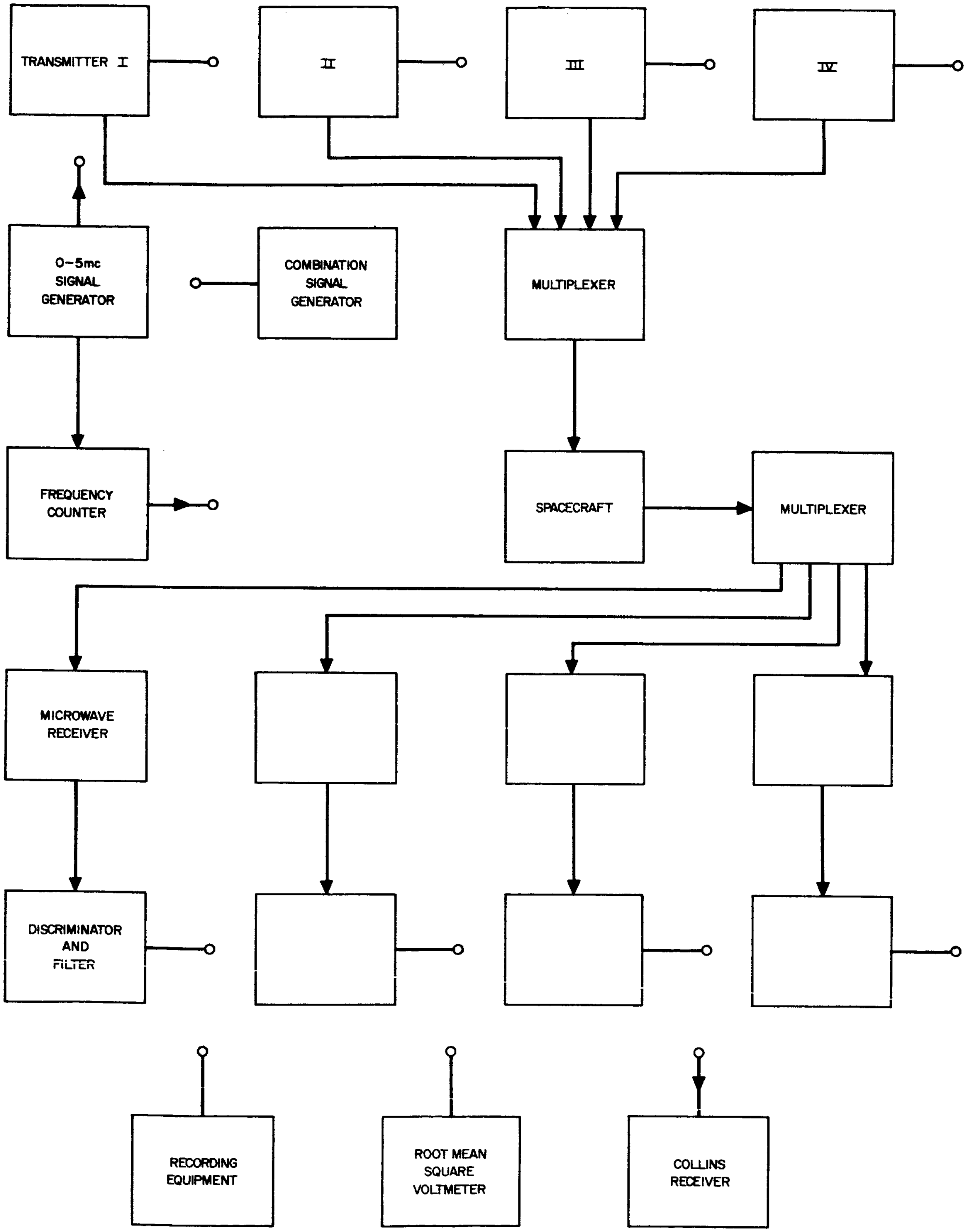
The test signal in the above tests will be filtered by use of a Collins Radio Model 51S single side band receiver which is used as a narrow-band variable filter.

Equipment Required

1) Microwave signal generator (multiple-access mode)
2) Combination signal generator
3) Calibrated discriminator
4) Spectrum analyzer
5) 0-5 mc signal generator
6) RMS voltmeter
7) Collins Radio Model 51S receiver

Spacecraft Access Required

1) RF input
2) RF output


TRANSMITTER I
II
III
IV
0–5mc SIGNAL GENERATOR
COMBINATION SIGNAL GENERATOR
MULTIPLEXER
FREQUENCY COUNTER
SPACECRAFT
MULTIPLEXER
MICROWAVE RECEIVER
DISCRIMINATOR AND FILTER
RECORDING EQUIPMENT
ROOT MEAN SQUARE VOLTMETER
COLLINS RECEIVER

MULTIPLE-ACCESS MODE TEST - TEST 13

Purpose of Test

To determine the intermodulation distortion of multiple-access transponder by measuring the noise power in received slot which is transmitted noise free.

Source of Test

Hughes, demonstrate spacecraft compatibility to CCIR Standards.

Test Procedure

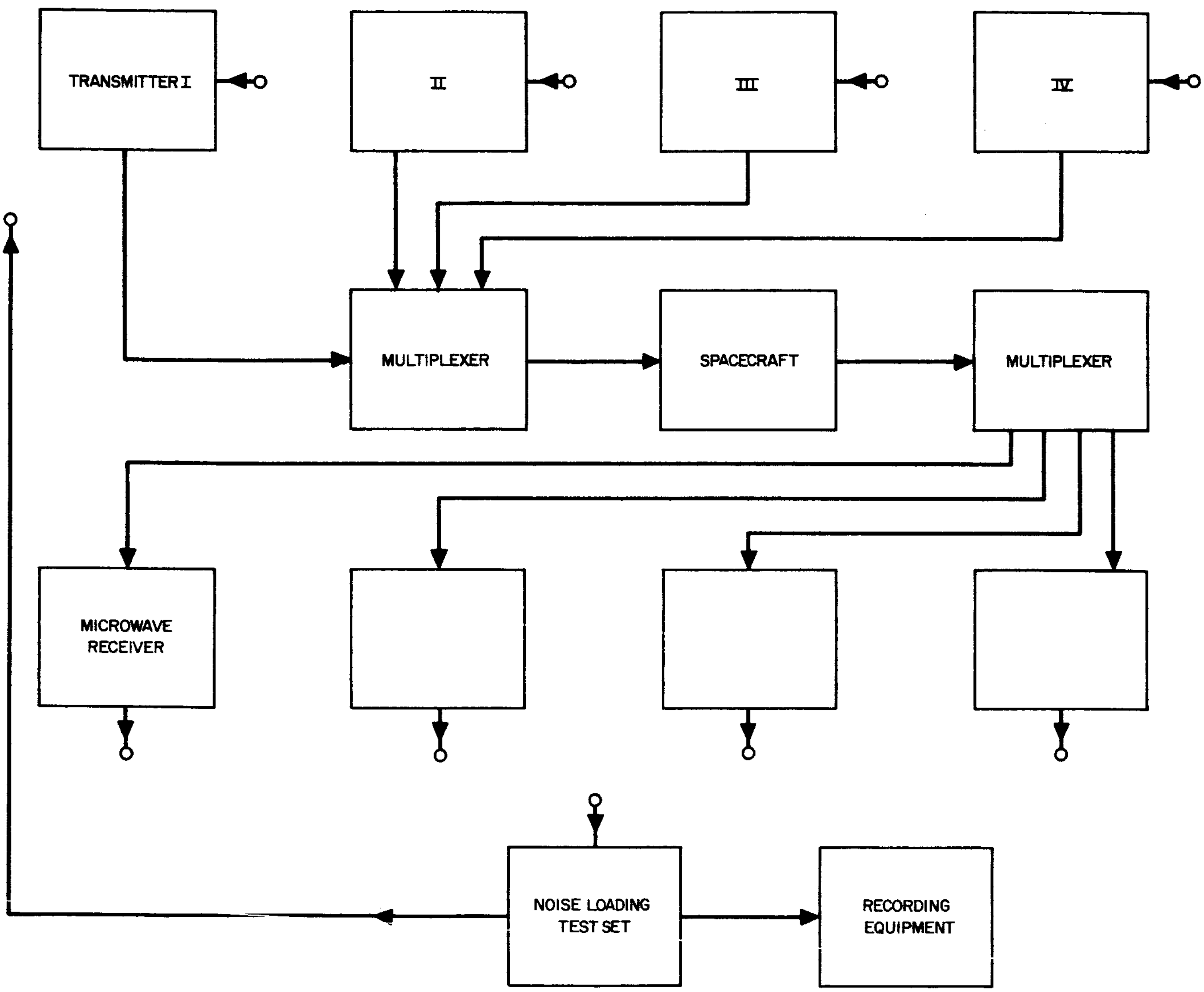
The Marconi White Noise test set will be used to generate a white noise base band which will be transmitted to the spacecraft via the test equipment dual-mode transmitter. Noise-free slots will be inserted in the transmitted base band at several frequencies. At the receiving end, the noise power introduced into the slot will be a measure of the total intermodulation produced in the base band signal. The support equipment will be calibrated so that its contribution to the intermodulation products is known.

Equipment Required

1) Microwave receiver
2) Microwave transmitter
3) Noise loading test set

Spacecraft Access Required

1) RF input
2) RF output


TRANSMITTER I
II
III
IV
MULTIPLEXER
SPACECRAFT
MULTIPLEXER
MICROWAVE RECEIVER
NOISE LOADING TEST SET
RECORDING EQUIPMENT

MULTIPLE-ACCESS TRANSPONDER - TEST 14

Purpose of Test

The output spectrum of multiple-access transponder will be examined for spurious output.

Source of Test

Hughes, demonstrate system performance

Test Procedure

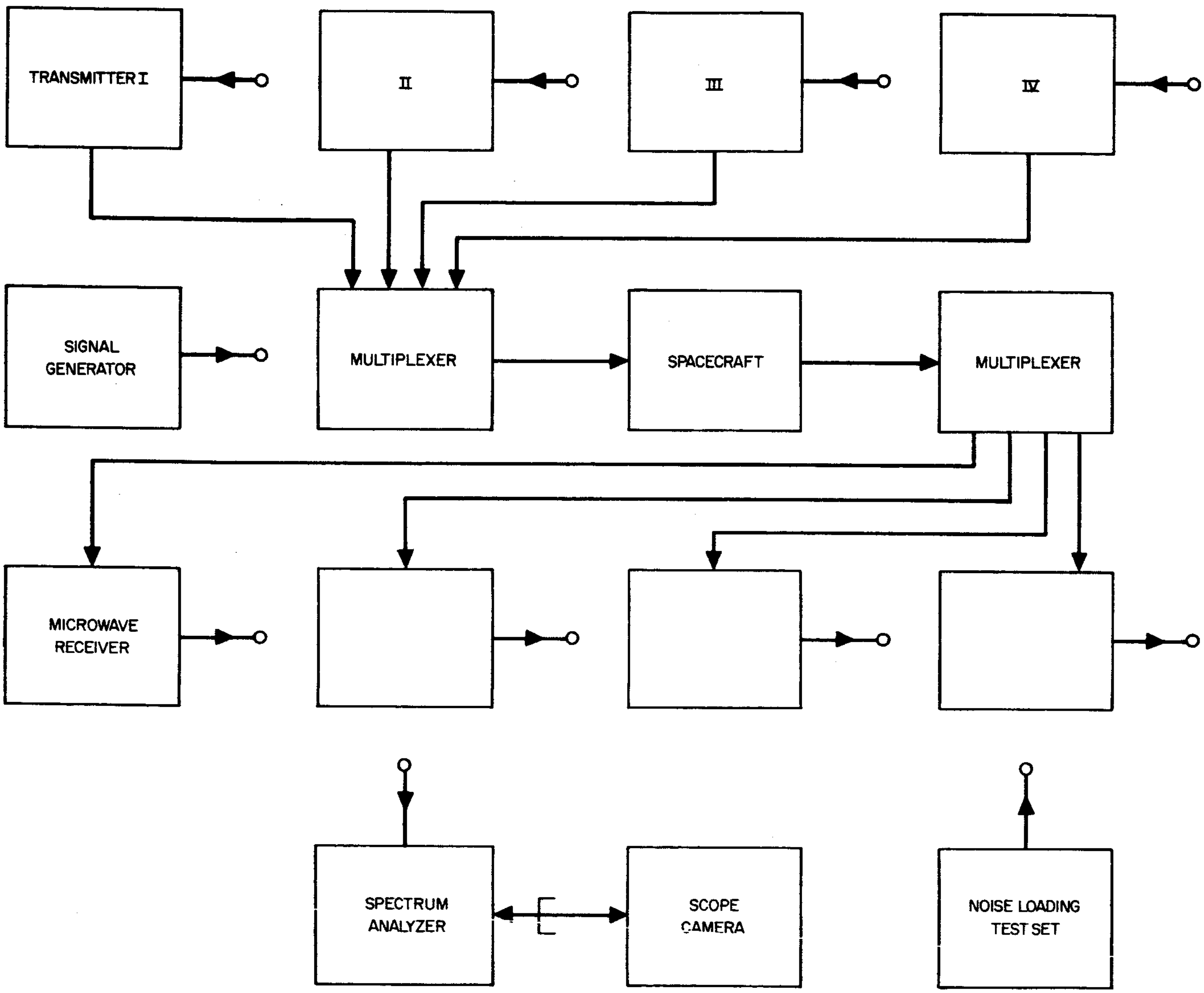
The four transponders will be turned on in the multiple-access mode, and the output spectrum of each will be examined for spurious output using a spectrum analyzer. A photograph record will be made of each spectrum.

Equipment Required

1) Spectrum analyzer
2) Camera
3) Signal generator
4) Microwave generator
5) Microwave receiver

Spacecraft Access Required

1) RF output
2) RF input


TRANSMITTER I
II
III
IV
SIGNAL GENERATOR
MULTIPLEXER
SPACECRAFT
MULTIPLEXER
MICROWAVE RECEIVER
SPECTRUM ANALYZER
SCOPE CAMERA
NOISE LOADING TEST SET

PHASED-ARRAY ANTENNA - TEST 15

Purpose of Test

Determine the stability and accuracy of the PACE Frequency-Locked Loop (FLL).

Source of Test

Hughes, to ensure proper operation of the PACE subsystem.

Test Procedure

The test will be performed with the spacecraft mounted on a spin machine and spun at normal spin speed. The spin machine used in this and successive phased-array tests will be specially designed for these tests. The machine will have sufficient stability to allow precise tests to be made. A light source which will be used to excite the solar sensors will be the reference for the spin machine.

The test signals will be brought from the spacecraft to the test equipment through slip rings.

The VCO of FLL, when properly locked, should oscillate at a frequency of 512 cycles per spacecraft revolution. A test access will make a sample of the VCO output available at the spacecraft test plug. A second signal, available at the spacecraft test plug, will be an output from the digital electronics of FLL that produces a pulse in synchronism with the ψ pulse when the loop is locked.

This test will be performed by counting the number of cycles out of the VCO between F-100 output pulses.

Equipment Required

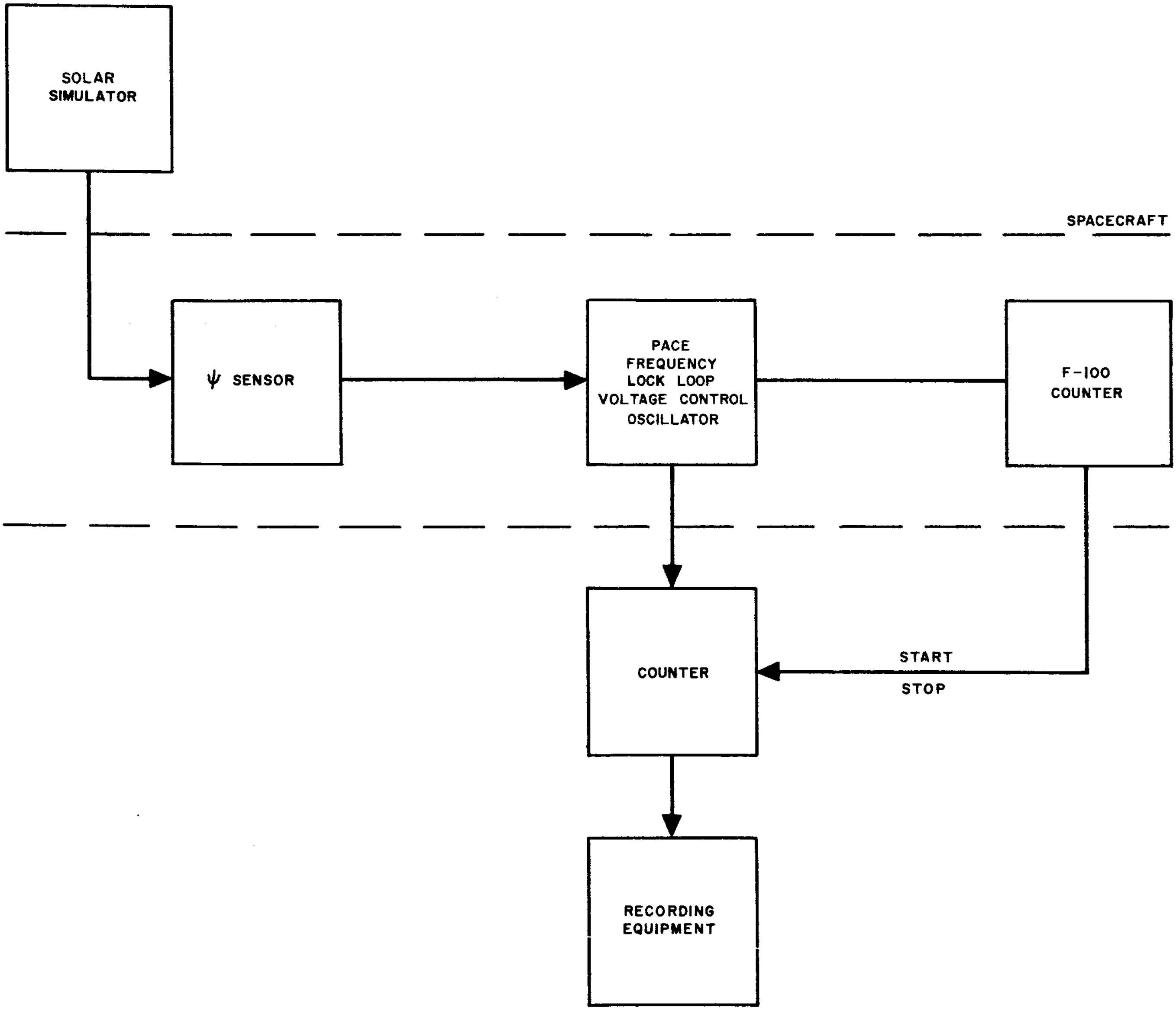
1) Solar simulator
2) Frequency counter
3) Recording equipment
4) Spin machine

Spacecraft Access Required

1) Frequency-locked loop VCO output
2) F-100 output

Spacecraft Access Required

1) Sample of PACE VCO output
2) Sample of F-100 output



PHASED-ARRAY ANTENNA - TEST 16

Purpose of Test

Determine the positioning accuracy and stability of the transmitted beam focused by the phased-array antenna.

Source of Test

Hughes, to ensure proper operation of the PACE subsystem.

Test Procedure

This test will be performed with the spacecraft mounted on the spin machine described in Test 15.

The position of the antenna beam, relative to the ψ source, will be determined by placing two horns in the field of the antenna and noting the difference in signal power from each horn. If the two horns are located such that they are equidistant from the center of the beam, then output from each horn will be equal, and therefore the output of the mixer will be zero.

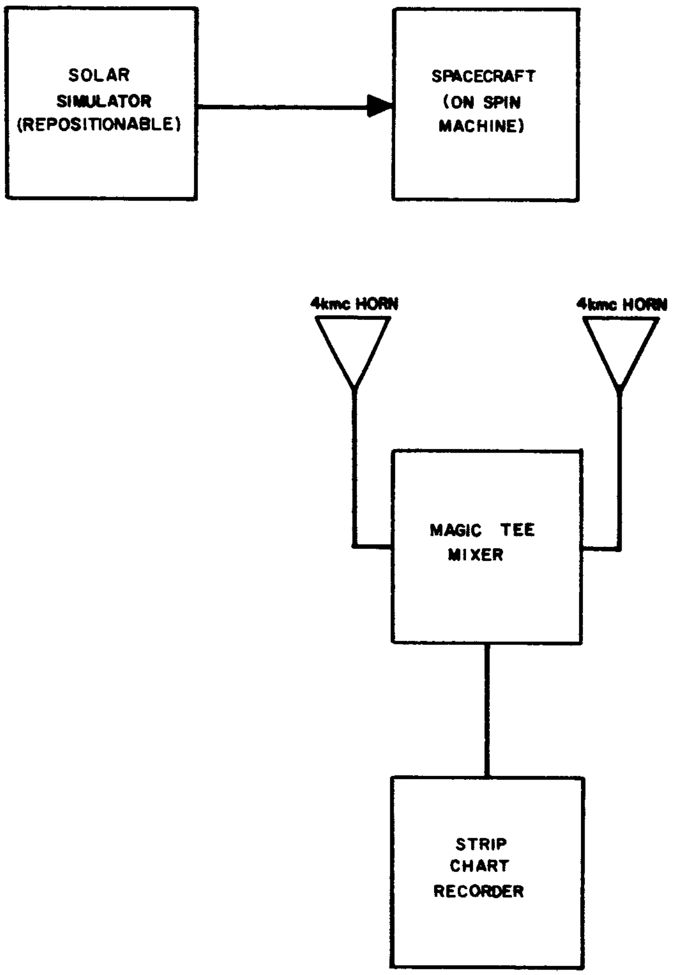
As the spacecraft spins, instability or asynchronism in the beam despin circuitry will cause the beam to have some apparent motion and therefore cause an output from the mixer.

The test fixtures will be calibrated by moving the beam off center and determining the mixer output as a function of degrees of beam mis-alignment. The mixer output will be recorded on a strip chart and the maximum beam drift during each cycle will be determined from maximum voltage out of the mixer.

The beam positioning accuracy will be determined by placing the solar simulator (four illuminators) at known angular positions around the periphery of the spacecraft and then commanding the beam such that its center should be between the field sensing horns.

Equipment Required

1) Four kmc horns (two each)
2) Magic tee mixer
3) 4-kmc mixer
4) Strip chart recorder
5) Spin machine


SOLAR
SIMULATOR
(REPOSITIONABLE)
SPACECRAFT
(ON SPIN
MACHINE)
4kmc HORN
4kmc HORN
MAGIC TEE
MIXER
STRIP
CHART
RECORDER

CENTRAL TIMER - TEST 17

Purpose of Test

To ensure the proper functioning of the central timer.

Source of Test

Hughes, to ensure spacecraft system operation.

Test Procedure

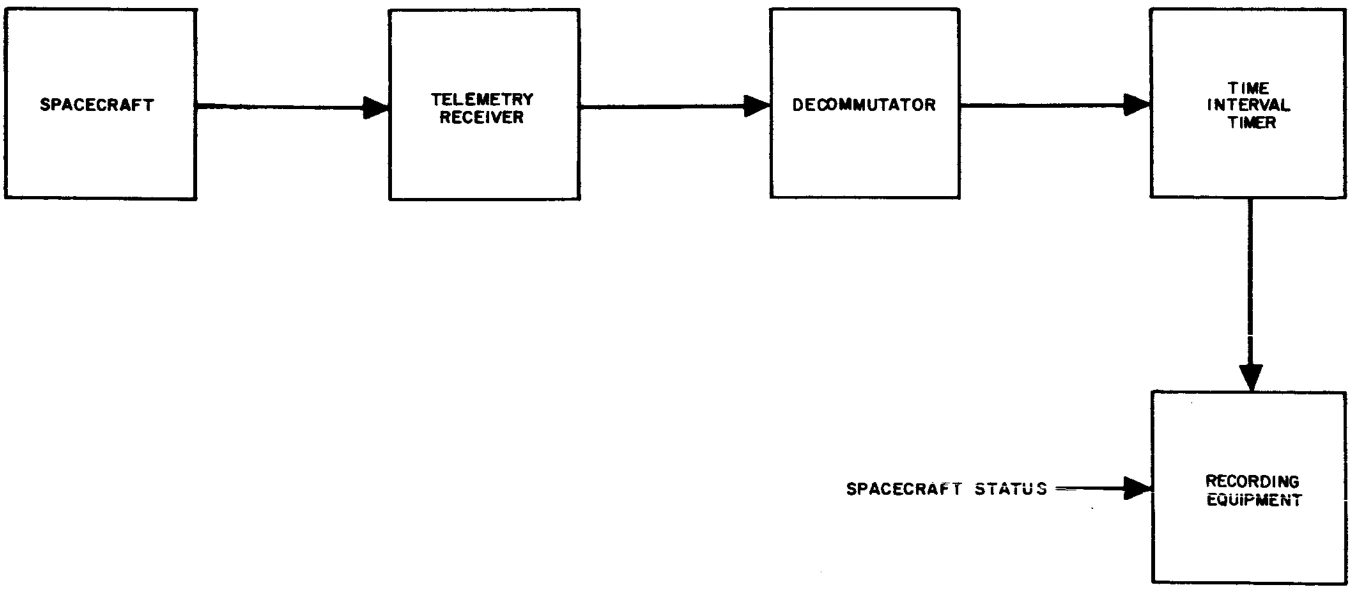
The telemetry frame rate will be directly controlled by the central timer. The telemetry processor will furnish a calibrate pulse during each frame. By measuring the time interval between calibration pulses of the decommutated telemetry signal, the frequency of the fork oscillator in the central timer may be determined. Each of the four timers will be tested in this manner.

Equipment Required

1) Telemetry receiver
2) Decommutator
3) Time interval counter

Spacecraft Access Required

None



APOGEE MOTOR IGNITOR CIRCUITRY - TEST 18

Purpose of Test

To ensure that the apogee motor ignitor circuitry is performing correctly.

Source of Test

Hughes, to ensure spacecraft system operation.

Test Procedure

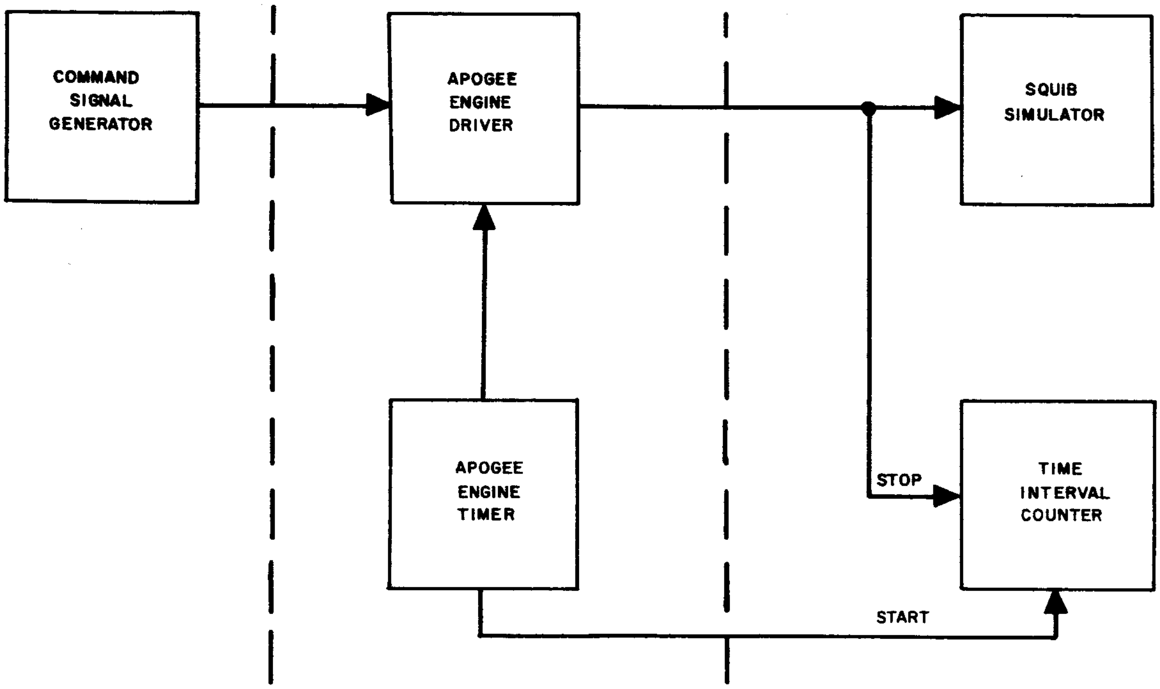
The apogee motor squib will be disconnected and a squib simulator, which will have similar electrical characteristics, will be placed in the circuit. The command to fire the apogee motor will then be generated and the simulator will be monitored for proper operation. The squib simulator will also be used to check the timing and firing circuitry in its complete long time form. The apogee timer will be allowed to run and the time from start to fire will be monitored, utilizing the squib simulator and a start-stop clock circuit.

Equipment Required

1) Command signal generator
2) Apogee motor squib simulator
3) Electric clock and start-stop circuitry

Spacecraft Access Required

1) Input to each pyrotechnic switch


COMMAND
SIGNAL
GENERATOR
APOGEE
ENGINE
DRIVER
SQUIB
SIMULATOR
APOGEE
ENGINE
TIMER
STOP
TIME
INTERVAL
COUNTER
START

APOGEE TIMER - TEST 19

Purpose of Test

To ensure the proper functioning of the apogee engine timer.

Source of Test

Hughes, to ensure spacecraft system operation.

Test Procedure

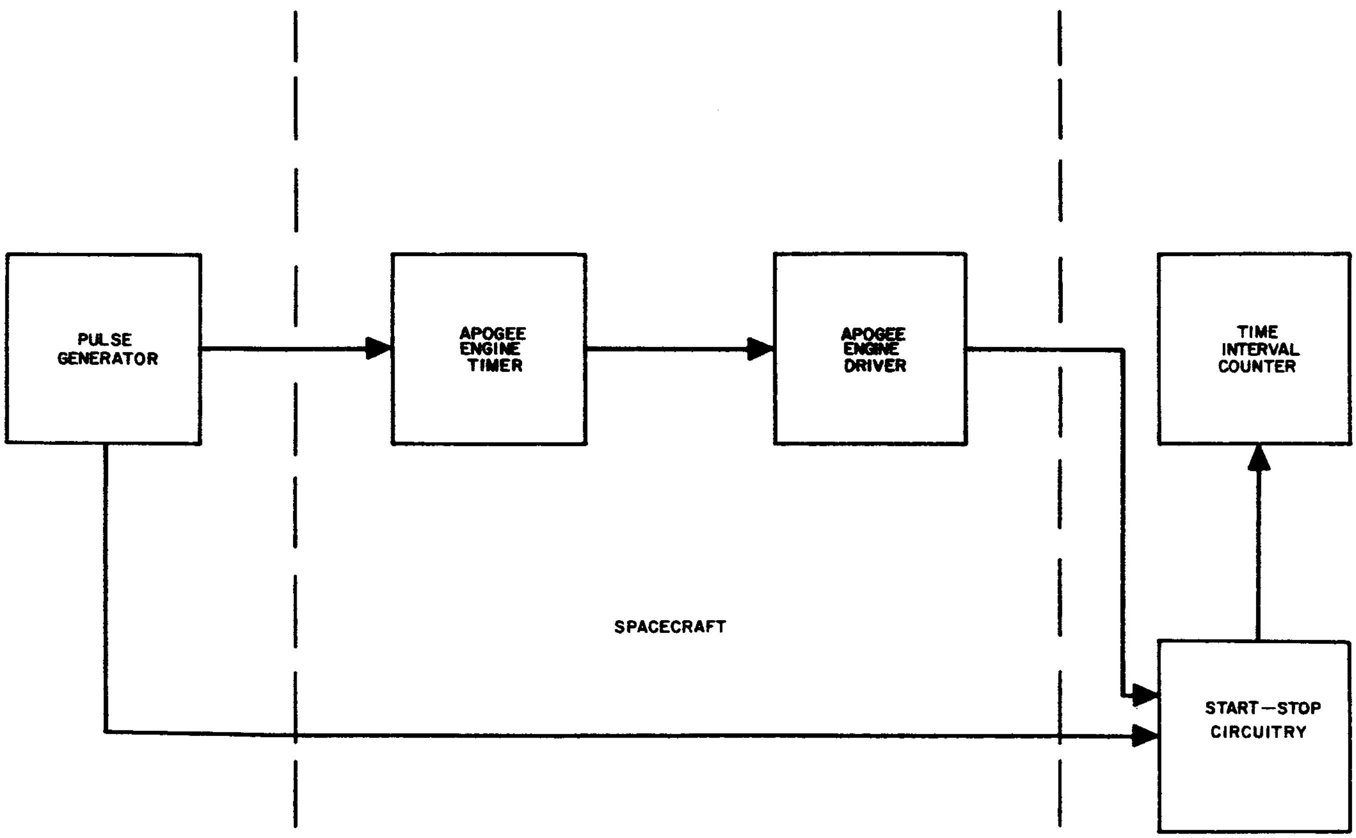
The apogee engine timer will employ saturable cores which receive pulses from the central timer, store the pulses until a predetermined number are present, and then feed the apogee engine driver. Four apogee engine timers will be used in the spacecraft, any combination of two being necessary to obtain an apogee engine fire signal. A variable frequency pulse generator will be used to rapidly pulse two timers simultaneously. By measuring the number of pulses required to obtain a fire signal, the apogee engine timer may be evaluated. In order to check each individual counter, one counter of the working pair will have a number of pulses fed into it which will allow it to be sufficiently ahead of the other timer to eliminate it from the test. All four timers will be evaluated in this manner.

Equipment Required

1) Electronic counter
2) Variable frequency pulse generator capable of stepping the apogee engine timer at a faster rate than normal

Spacecraft Access Required

1) Apogee timer input
2) Apogee motor fire signal


PULSE GENERATOR
APOGEE ENGINE TIMER
APOGEE ENGINE DRIVER
TIME INTERVAL COUNTER
SPACECRAFT
START—STOP CIRCUITRY

SOLAR SENSORS - TEST 20

Purpose of Test

To monitor the solar sensors for proper operation.

Source of Test

Hughes, to ensure system operation

Test Procedure

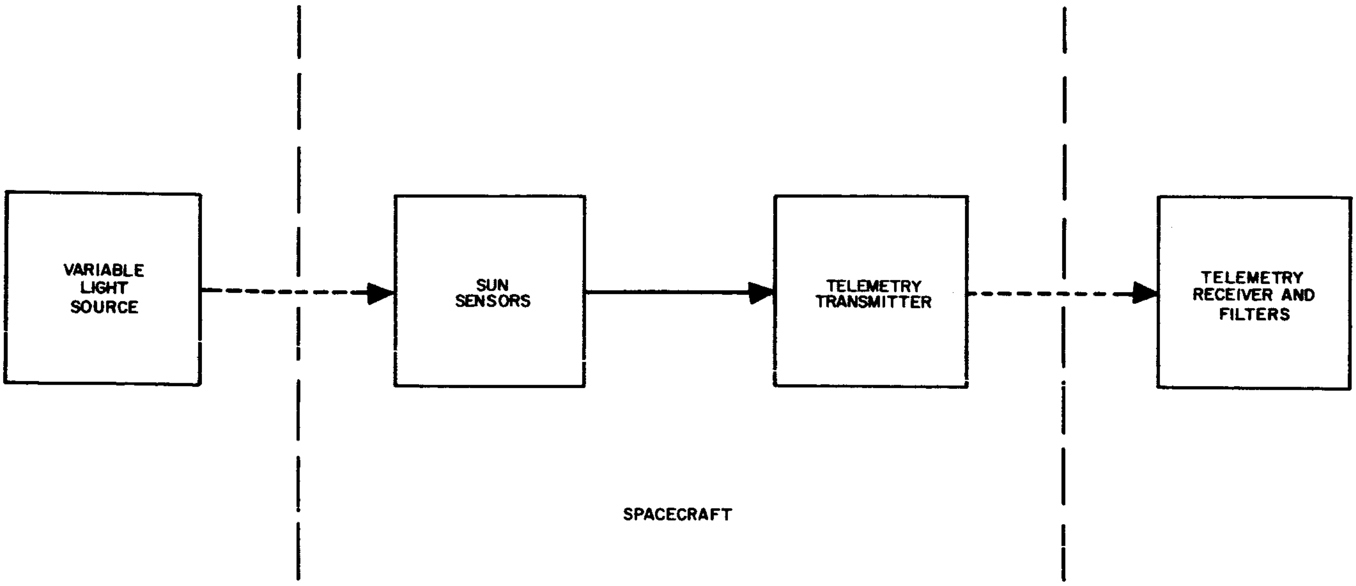
The solar sensor pulses will be monitored at the output of the telemetry receiver - solar pip filter for their presence as the spacecraft is spun past a solar simulator. When a normal earth reflection level of light is on the solar sensor, no solar pip should appear from the output of the telemetry receiver.

Equipment Required

1) Solar simulator light source whose intensity may be varied
2) Telemetry receiver and solar pip filter

Spacecraft Access Required

None



ORIENTATION AND CONTROL SUBSYSTEM - TEST 21

Purpose of Test

Determine the accuracy of the telemetry $\psi - \psi_2$ angle encoder.

Source of Test

Hughes, to ensure proper system operation

Test Procedure

The spacecraft will be mounted on the spin fixture and spun at a normal speed. The ψ and ψ_2 sensors will be illuminated by a solar simulator.

The transmitted ψ and ψ_2 pulses will be received by the support equipment. The telemetry receiver demodulates the ψ and ψ_2 pulses and applies them to the backup ground synchronous control equipment. The $\psi - \psi_2$ angle will be measured and compared with angle encoded and transmitted via the telemetry subsystem. The four illuminators will then be repositioned and the new angle checked.

An alternate method of performing the test is to use a time interval counter to measure the time between the occurrence of ψ and ψ_2. Then:

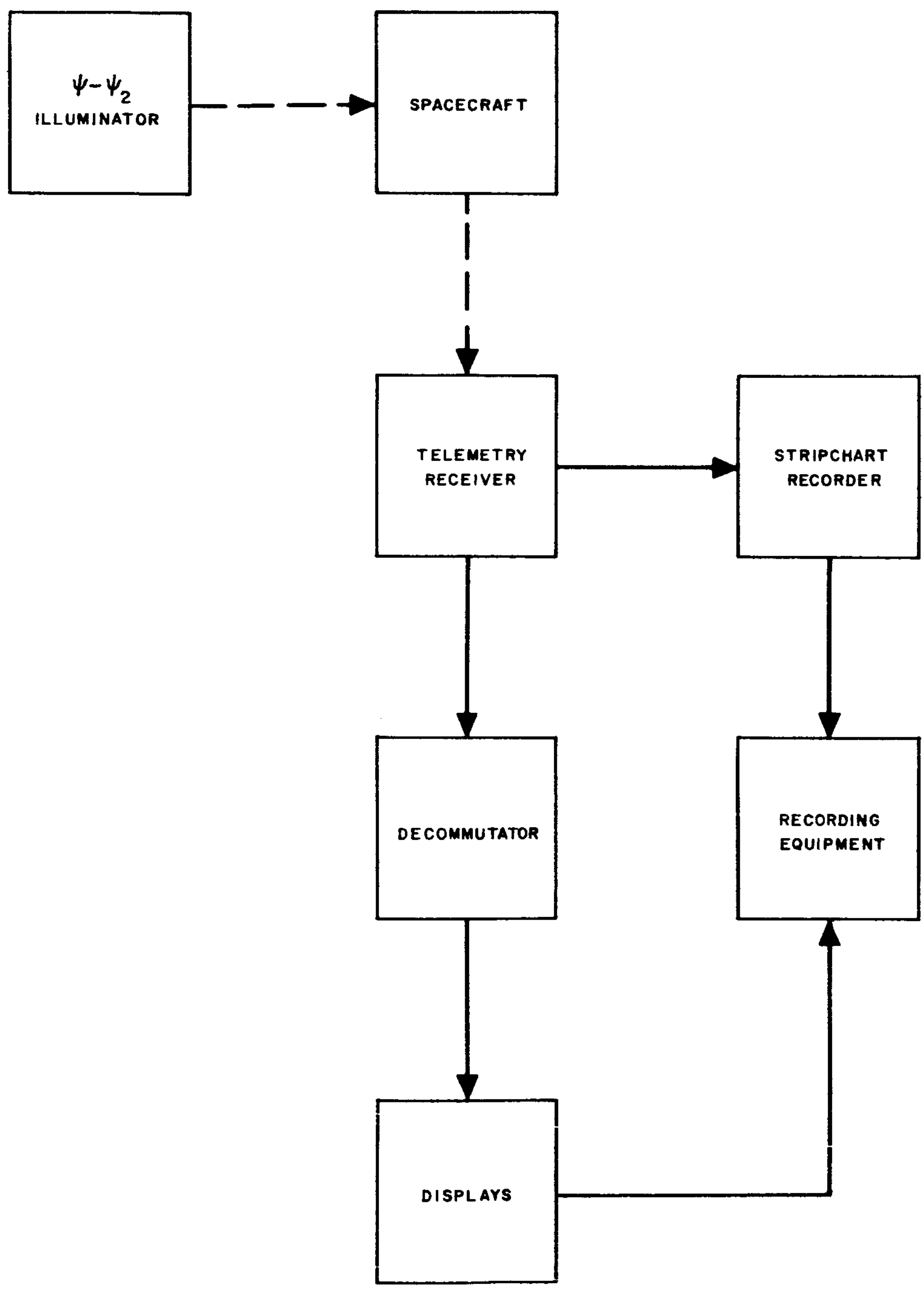
$$\psi - \psi_2 \text{ in degrees } = \left(\frac{\psi - \psi_2 \text{ in seconds}}{\text{rev /sec}} \right) (360)$$

Equipment Required

1) Four illuminators
2) Ground synchronous controller
3) TM receiver
4) Spin machine

Spacecraft Access Required

None



$\psi - \psi_2$
ILLUMINATOR
SPACECRAFT
TELEMETRY RECEIVER
STRIPCHART RECORDER
DECOMMUTATOR
RECORDING EQUIPMENT
DISPLAYS

ORIENTATION AND CONTROL SUBSYSTEM - TEST 22

Purpose of Test

Determine the angle, relative to the antenna beam center, at which a commanded jet fires. Determine the angle through which the jet fires.

Source of Test

Hughes, to ensure proper system operation.

Test Procedure

The spacecraft will be mounted on the spin fixture and spun at normal speed.

When operating in its normal mode the PACE electronics will, on command, fire a selected jet when it is ninety (90) degrees from the center of the antenna beam. This test will measure the angle at which the jet is fired relative to the ψ pulse, through use of the backup ground synchronous controller. With the ground synchronous controller in its "time delay" mode, the spin angle between ψ and jet fire, as indicated by a change in execute tone amplitude, can be measured directly. The angle is then compared to the angle predicted by the following:

Antenna beam angle + or - (90 degree delay constants)

= angle measured

The angle through which the jet fires will be measured by the difference in spin between the start of jet fire (as above) and the conclusion of jet fire as indicated by the execute tone returning to its original amplitude.

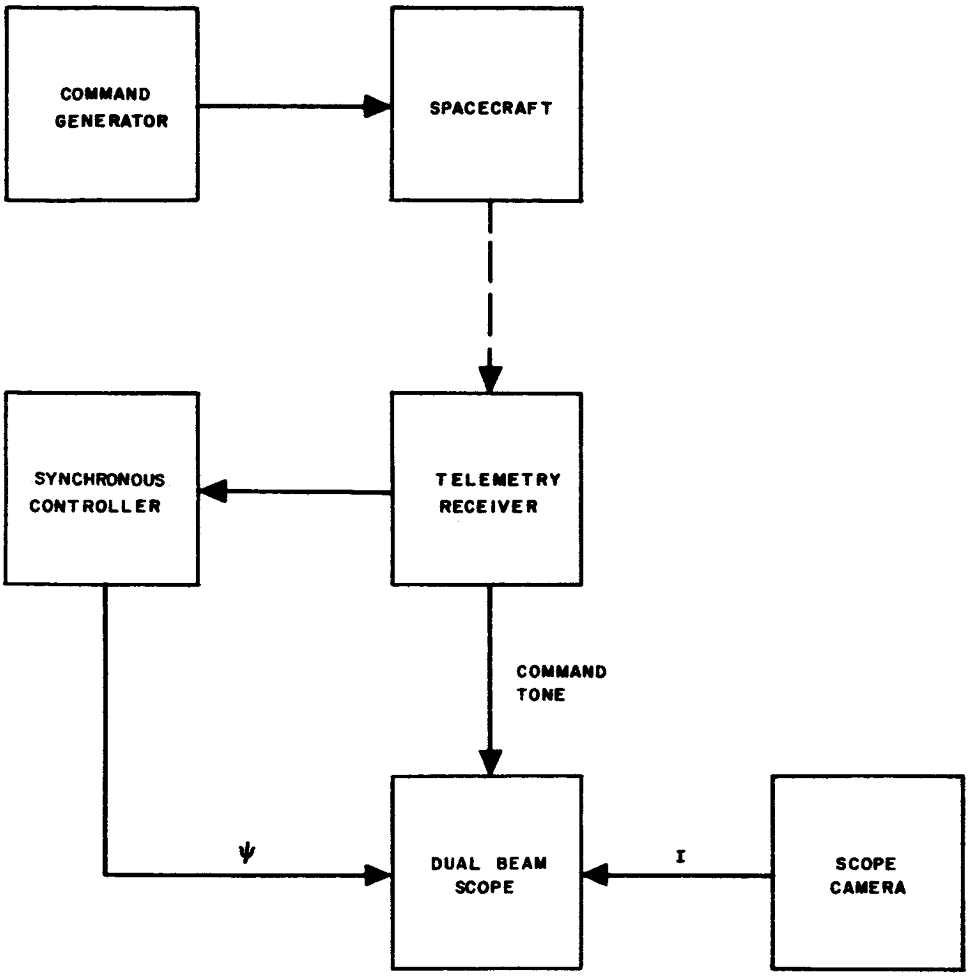
An alternate method of performing this test would be to measure the time interval between ψ and the execute tone indication of jet fire.

Equipment Required

1) Backup ground synchronous controller
2) ψ sensor illuminator
3) TM receiver
4) Spin machine

Spacecraft Access Required

None


COMMAND GENERATOR
SPACECRAFT
SYNCHRONOUS CONTROLLER
TELEMETRY RECEIVER
COMMAND TONE
ψ
DUAL BEAM SCOPE
I
SCOPE CAMERA

BIPROPELLANT SQUIBS - TEST 23

Purpose of Test

To test the bipropellant squib circuitry for proper operation.

Source of Test

Hughes, to ensure spacecraft system operation.

Test Procedure

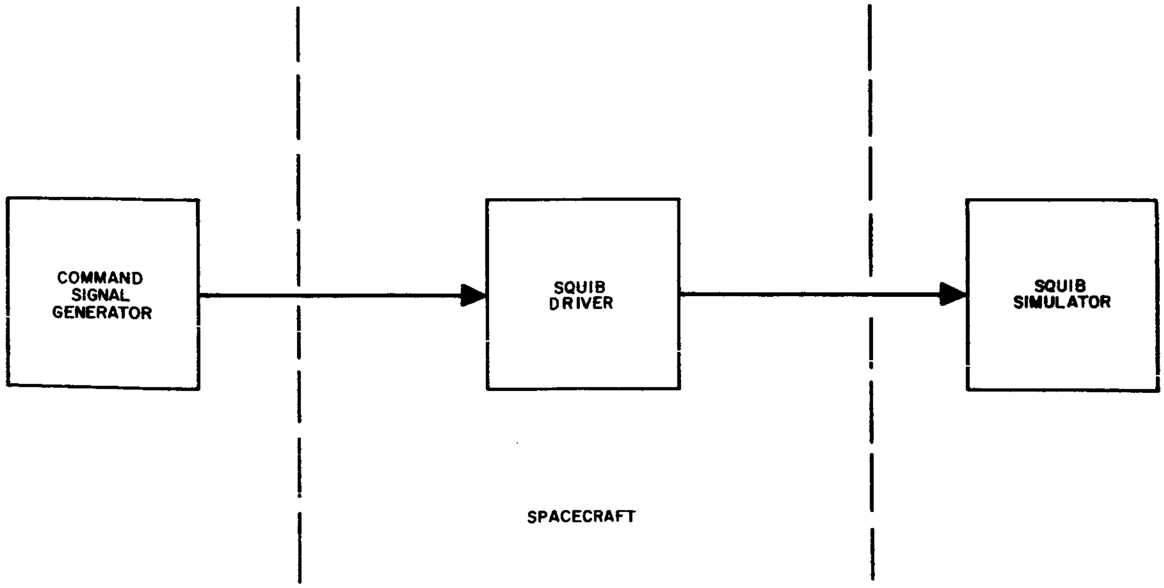
The bipropellant squibs will be disconnected and a squib simulator, which will have similar electrical characteristics, will be placed in the circuit. The command to fire the bipropellant squibs will then be generated and the simulator will be monitored for proper operation. The test will be repeated for each squib.

Equipment Required

1) Command signal generator
2) Bipropellant squib simulator

Spacecraft Access Required

Input to each pyrotechnic squib



SEPARATION SWITCHES - TEST 24

Purpose of Test

To ensure that the separation switches are functioning correctly.

Source of Test

Hughes, to ensure system operation.

Test Procedure

It is assumed that the function of the separation switches will be to:

a) Hold the pyrotechnic devices in a safe configuration
b) Keep the apogee motor timer reset until separation

The pyrotechnic inputs will be tested by sending appropriate commands to the spacecraft and noting that no response is obtained as long as the separation switches are held in their energized position. The switches will then be released, appropriate command sent, and it will be noted that proper action occurs.

The apogee timer separation switches will be tested by noting that the apogee timer remains in reset condition as long as the switches are energized. With the switches de-energized, the apogee timer will be "speed-up tested" by determining the number of counts it takes to fire.

Equipment Required

Spacecraft Access Required

1) Pyrotechnic switch inputs
2) Input to apogee timer

COMMAND RECEIVER - TEST 25

Purpose of Test

Determine the frequency response of the command receiver.

Source of Test

Hughes, to ensure proper system performance.

Test Procedure

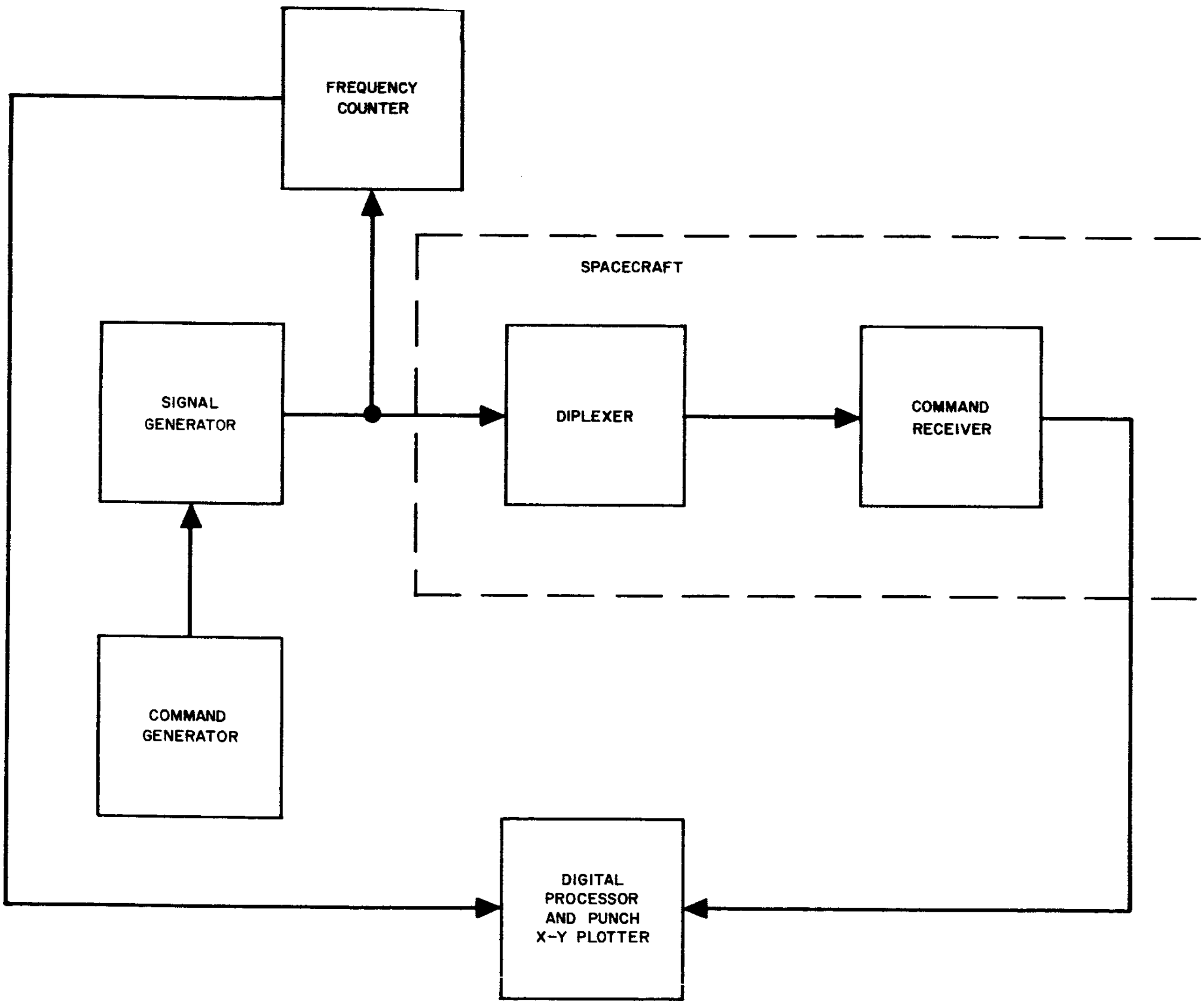
The RF frequency response of the command receiver will be determined by plotting the audio output as a function of RF input frequency. This test will be conducted with the RF input signal adjusted for a low level on the order of -100 dbm. The plot of the output audio will demonstrate that the receiver output is adequate over the prescribed bandwidth. The position of the audio bandpass relative to the frequency axis will demonstrate that the command receiver local oscillator is operative on the proper frequency.

Equipment Required

1) RF signal generator
2) Audio signal generator
3) RMS voltmeter
4) Frequency meter
5) Data recorder

Spacecraft Access Required

1) Command RF input
2) Command audio output


FREQUENCY
COUNTER
SPACECRAFT
SIGNAL
GENERATOR
DIPLEXER
COMMAND
RECEIVER
COMMAND
GENERATOR
DIGITAL
PROCESSOR
AND PUNCH
X-Y PLOTTER

COMMAND RECEIVER - TEST 26

Purpose of Test

Determine the sensitivity of the command subsystem.

Source of Test

NASA Specification S2-0100, paragraph 3.10.2.1.7.

Test Procedure

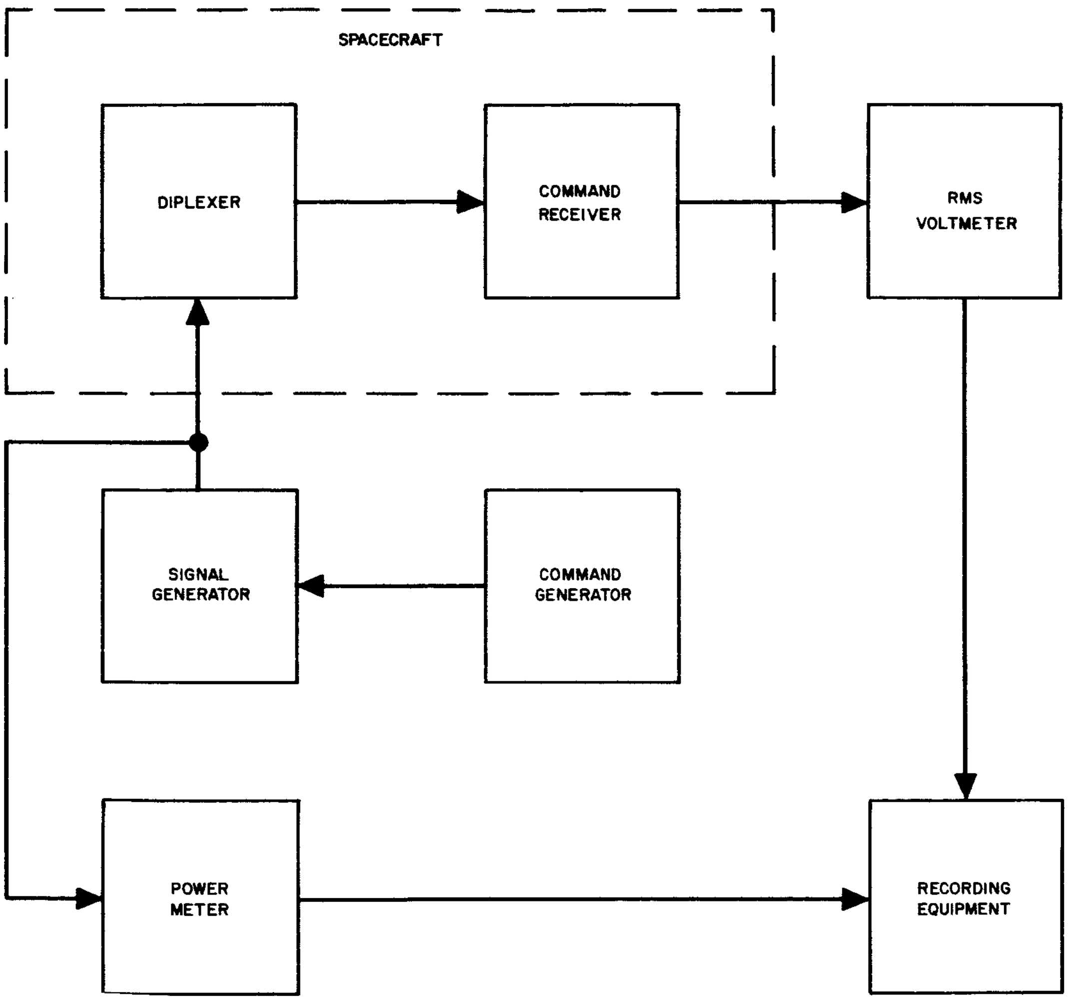
The center RF frequency sensitivity of the command receiver will be determined by plotting audio signal output as a function of RF signal input. This test will be performed by inserting a modulated RF signal into the command receiver and plotting the audio signal output of the command receiver test point. The audio signal frequency used will be close to the normal audio command tone frequency.

Equipment Required

1) Modulated RF signal generator
2) Power meter
3) RMS voltmeter
4) Recording equipment

Spacecraft Access Required

1) RF input to command receiver
2) Audio output of command receiver


SPACECRAFT
DIPLEXER
COMMAND RECEIVER
RMS VOLTMETER
SIGNAL GENERATOR
COMMAND GENERATOR
POWER METER
RECORDING EQUIPMENT

COMMAND DECODER - TEST 27

Purpose of Test

To ensure that the command system responds to all commands in both primary and backup modes.

Source of Test

Hughes, to ensure proper system operation and compatibility with its GSE.

Test Procedure

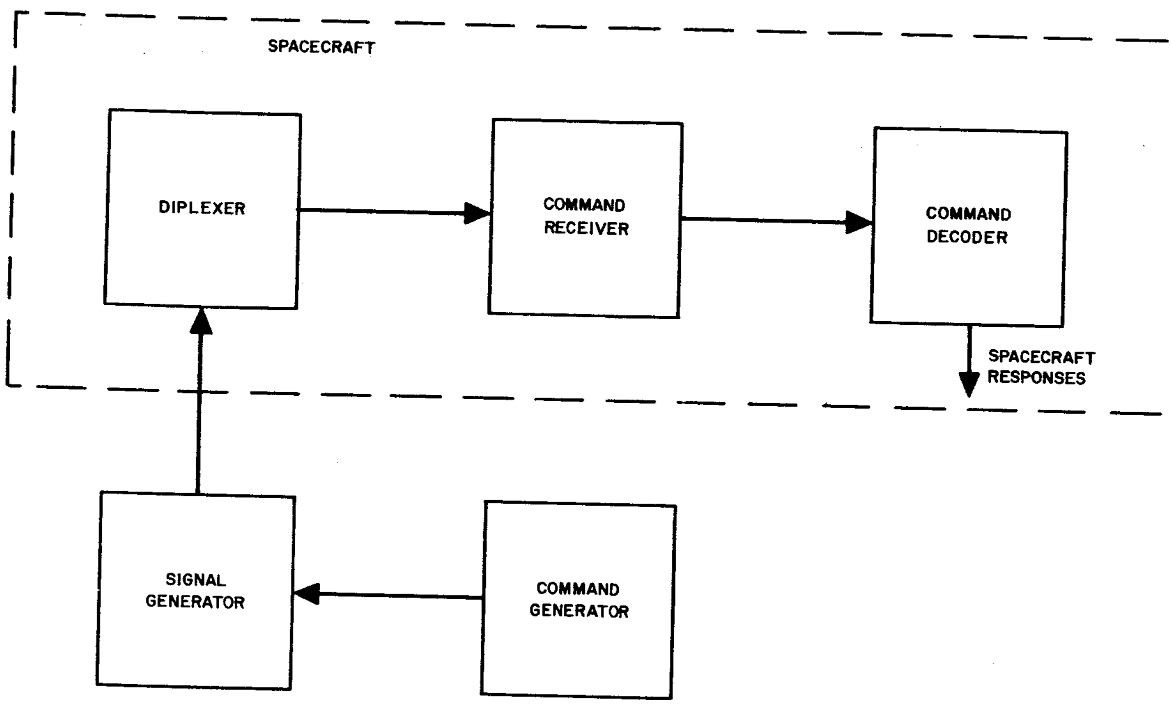
The support equipment command signal generator will be used to generate all spacecraft commands in the primary command mode. The spacecraft responses will be monitored with various pieces of equipment to ensure adequate spacecraft response. The test will be repeated with the command signal generator operating in the command system backup mode.

Equipment Required

1) Dual-mode command signal generator
2) Spectrum analyzer
3) RF signal generator
4) Telemetry decommutator and display

Spacecraft Access Required

1) Command RF input


SPACECRAFT
DIPLEXER
COMMAND RECEIVER
COMMAND DECODER
SPACECRAFT RESPONSES
SIGNAL GENERATOR
COMMAND GENERATOR

TELEMETRY SUBSYSTEM - TEST 28

Purpose of Test

Determine the frequency of each telemetry transmitter.

Source of Test

NASA Specification S2-0100, paragraph 3. 10. 1. 2. 1.

Test Procedure

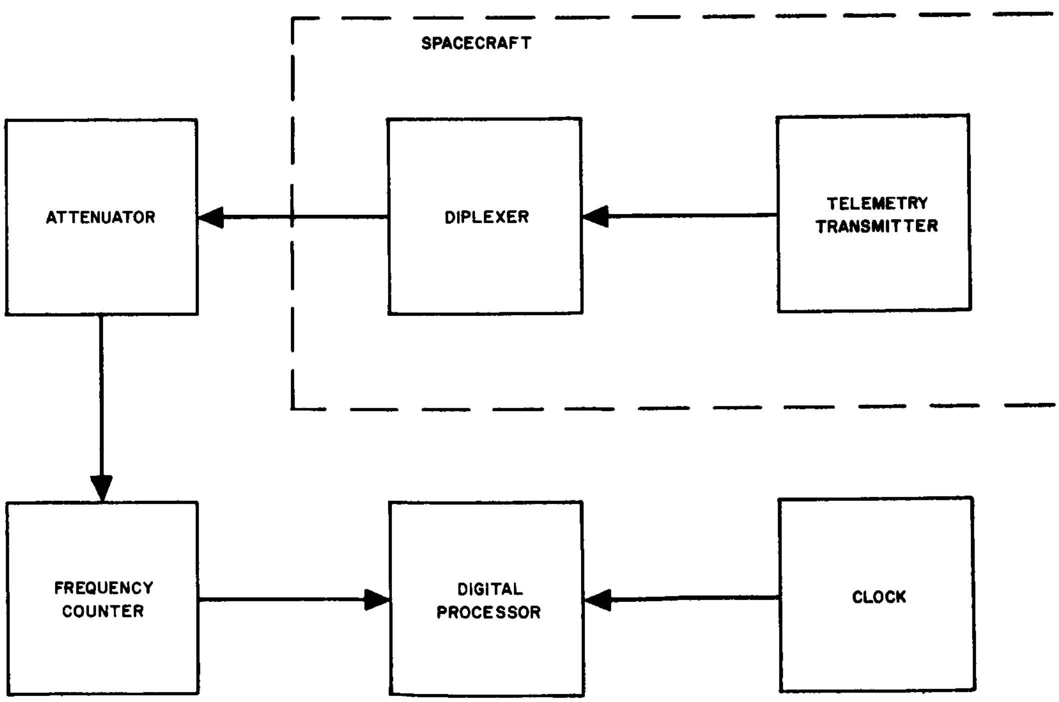
The frequency of the telemetry transmitter will be measured by applying the output telemetry transmitter RF test access directly to the frequency counter. The frequency stability will be determined by recording the frequency over a period of time.

Equipment Required

1) Frequency counter with 100-200 mc plug-in
2) Digital recorder

Spacecraft Access Required

1) TM RF output



TELEMETRY SUBSYSTEM - TEST 29

Purpose of Test

Determine the power output of the telemetry transmitter.

Source of Test

NASA Specification S2-0100, paragraph 3.10.1.2.3.

Test Procedure

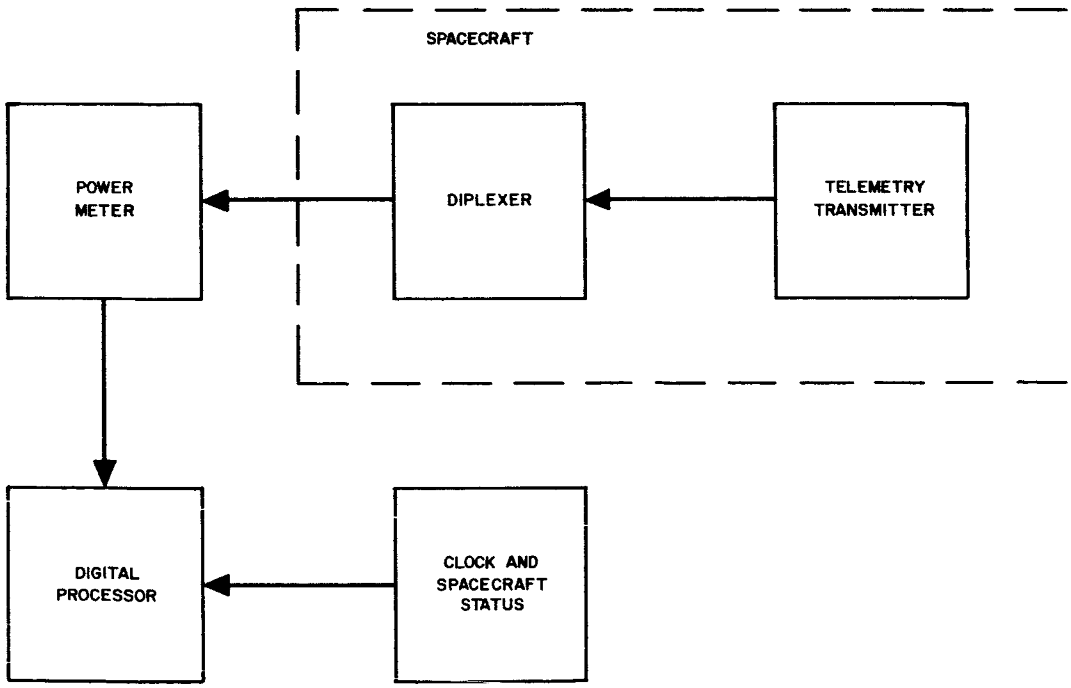
The power output of telemetry transmitter will be measured at the output of a directional coupler built into the spacecraft between the telemetry transmitter/command receiver diplexer and the T and C balun.

Equipment Required

1) Power meter

Spacecraft Access Required

1) T and C directional coupler output



TELEMETRY ENCODER - TEST 30

Purpose of Test

Determine the frequency range and linearity of encoder subcarrier oscillator.

Source of Test

Hughes, to ensure spacecraft operating parameters.

Test Procedure

The encoder frequency range will be determined by measuring the VCO frequency during the +5-volt and the 0-volt calibration channels. These two measurements will define a straight line on the VCO input voltage/output frequency plot and determine its frequency range.

The VCO linearity will be checked by measuring several input voltages to the VCO and comparing the actual frequency output caused by these voltages with the values predicted by the input voltage/frequency curve determined above.

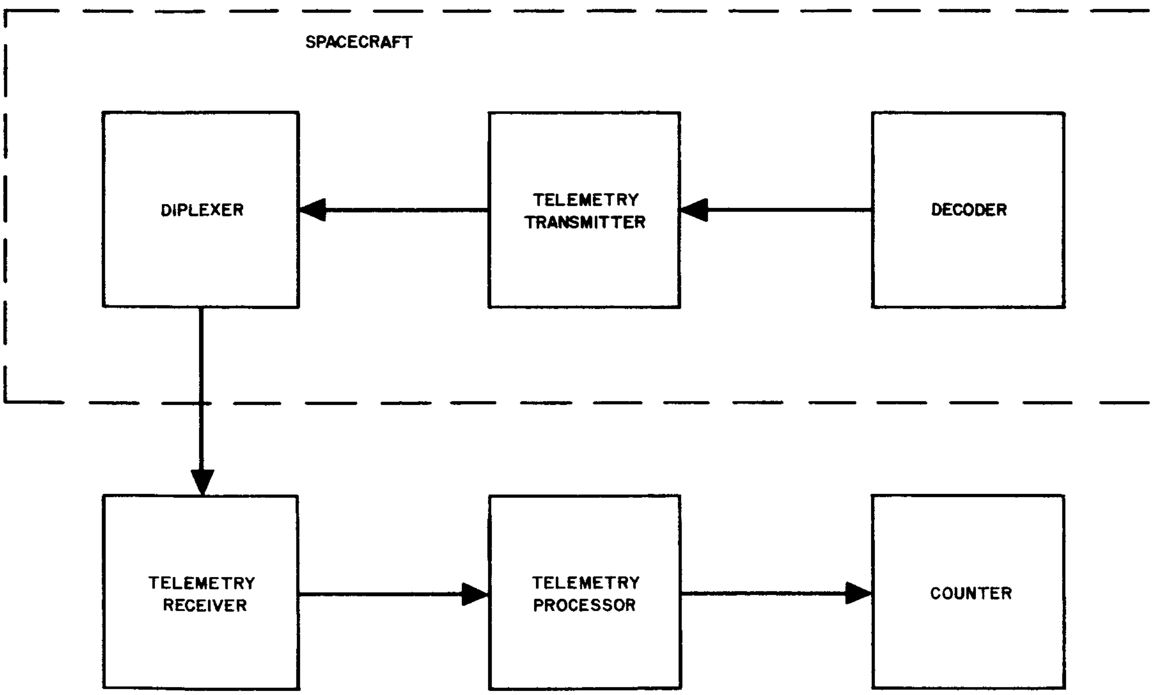
The telemetry processor will be used to measure the VCO frequency at the selected channels indicated in this test.

Equipment Required

1) TM receiver
2) TM data processor
3) Frequency counter

Spacecraft Access Required

1) RF input to diplexer
2) Battery voltage output


SPACECRAFT
DIPLEXER
TELEMETRY TRANSMITTER
DECODER
TELEMETRY RECEIVER
TELEMETRY PROCESSOR
COUNTER

TELEMETRY ENCODER - TEST 31

Purpose of Test

To ensure that the telemetry encoder is operating properly by checking the Telemetric frame for presence of all channels.

Source of Test

Hughes, to ensure proper system operation

Test Procedure

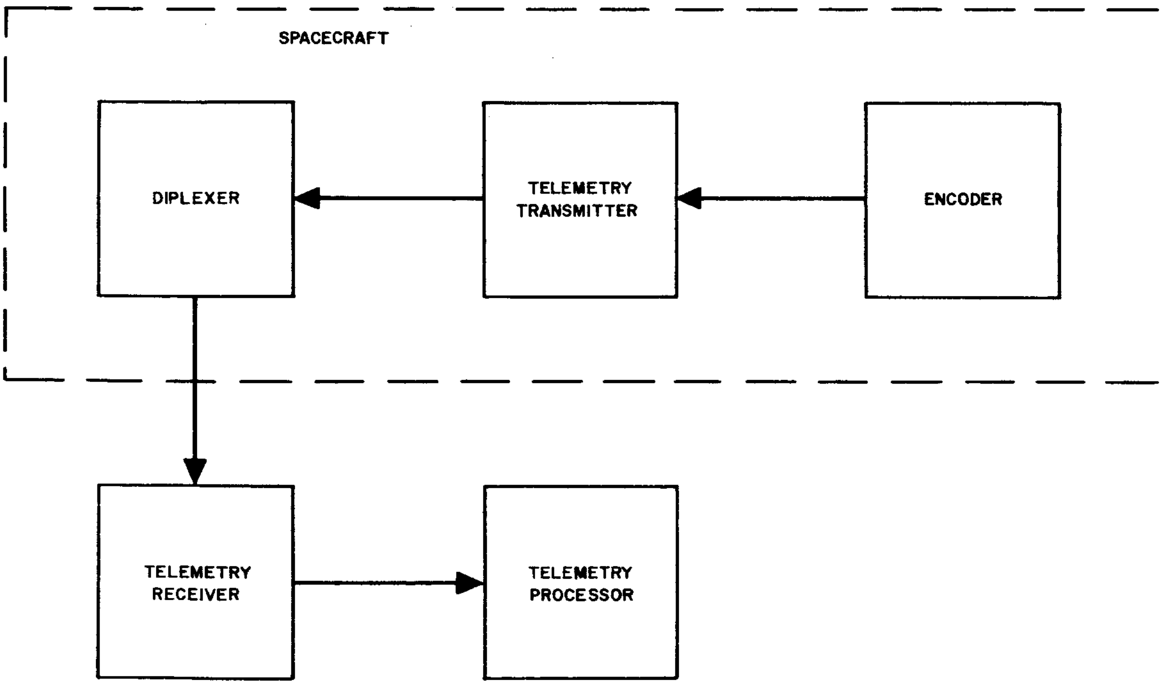
The output of the telemetry transmitter will be demodulated by the support equipment telemetry receiver. The TM data will then be processed by the decommutator. The output of the decommutator will be checked for the presence of all information.

Equipment Required

1) TM receiver
2) TM processor
3) Digital recorder

Spacecraft Access Required

1) TM output


SPACECRAFT
DIPLEXER
TELEMETRY TRANSMITTER
ENCODER
TELEMETRY RECEIVER
TELEMETRY PROCESSOR

SOLAR PANEL - TEST 32

Purpose of Test

To check the functional operation of the solar panels.

Source of Test

Hughes, to ensure system operations

Test Procedure

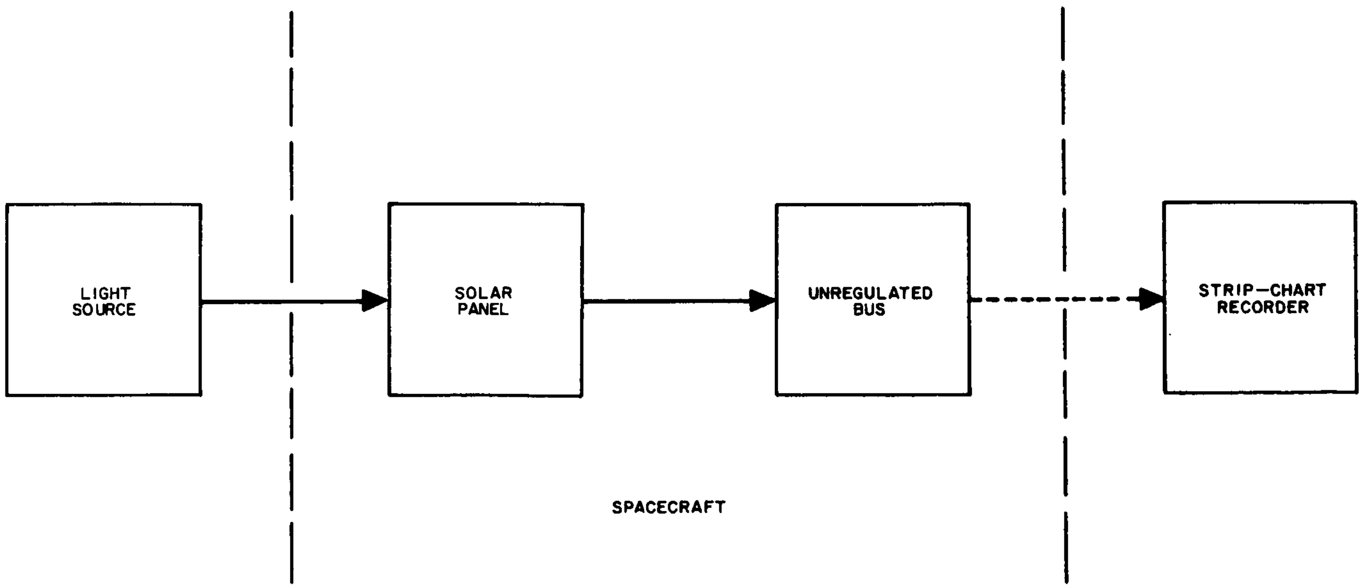
A light source capable of lighting one solar string at a time will be mounted on the spacecraft handling fixture. As the spacecraft is rotated at approximately 1 rpm, the unregulated bus voltage will be recorded on a stripchart recorder. A malfunctioning solar string will be detected by a drop in unregulated bus voltage as seen on the stripchart recorder.

Equipment Required

1) Stripchart recorder

Spacecraft Access Required

1) Unregulated bus



SUBSYSTEM POWER CONSUMPTION - TEST 33

Purpose of Test

To monitor the power consumption of each subsystem.

Source of Test

Hughes, to observe system performance.

Test Procedure

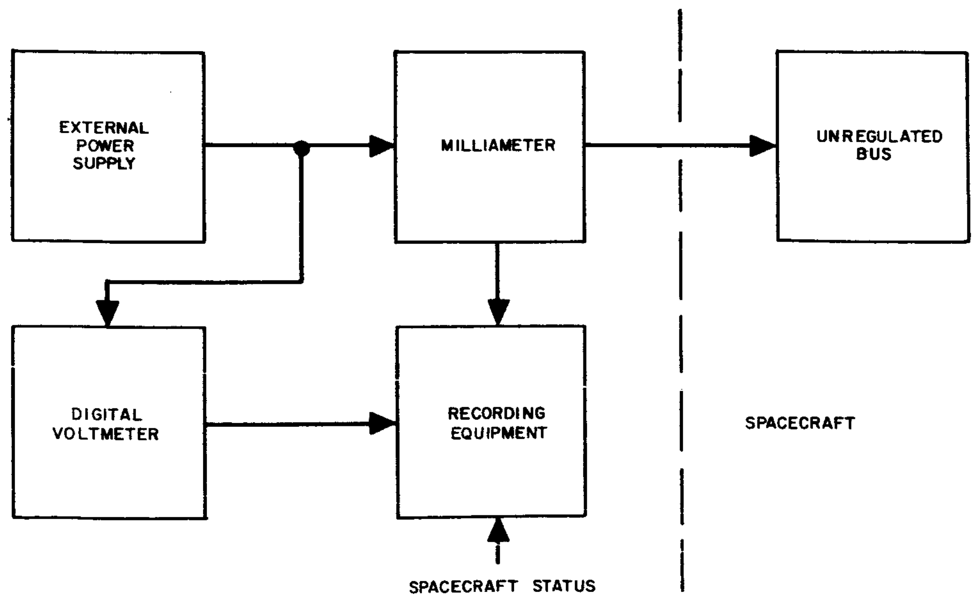
With the spacecraft power supplied from an external dc source possessing the same impedance as the solar panels, each subsystem will be turned on. The input voltage will be maintained at -32 volts and the current drawn will be monitored by a milliamp meter and digitally recorded.

Equipment Required

1) External power supply
2) Milliamp meter
3) Digital voltmeter
4) Recording equipment

Spacecraft Access Required

None



UNREGULATED BUS VOLTAGE - TEST 34

Purpose of Test

To determine the unregulated bus voltage characteristics as to voltage, ripple, and transients.

Source of Test

NASA Specification S2-0100, paragraphs 3.8.2.1, 3.8.2.2, and 3.8.2.3

Test Procedure

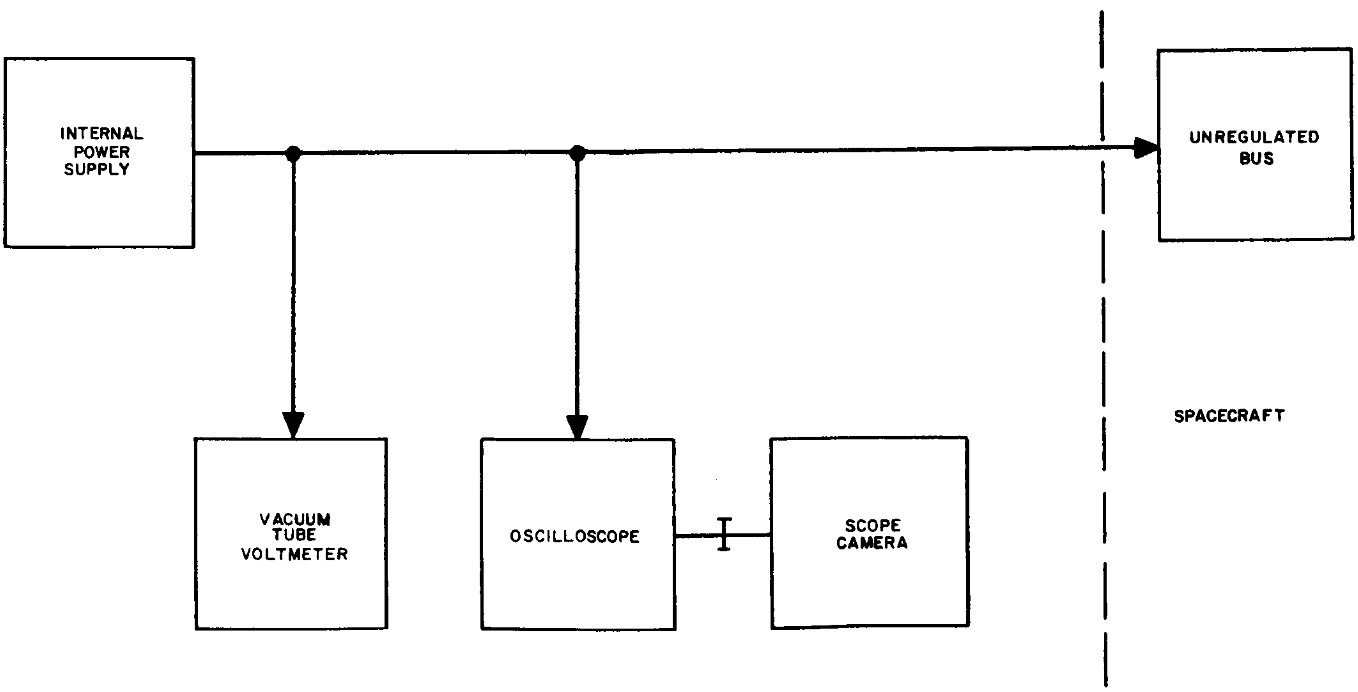
With the spacecraft operating on internal power, each TWT will be turned on and the unregulated bus will be monitored on a recording oscillograph.

Equipment Required

1) Oscilloscope
2) Oscilloscope camera

Spacecraft Access Required

1) Unregulated bus



REGULATOR TESTS - TEST 35

Purpose of Test

To ensure the proper operation of the subsystem regulators.

Source of Test

NASA Specification S2-0100, paragraph 3.8.5.2

Test Procedure

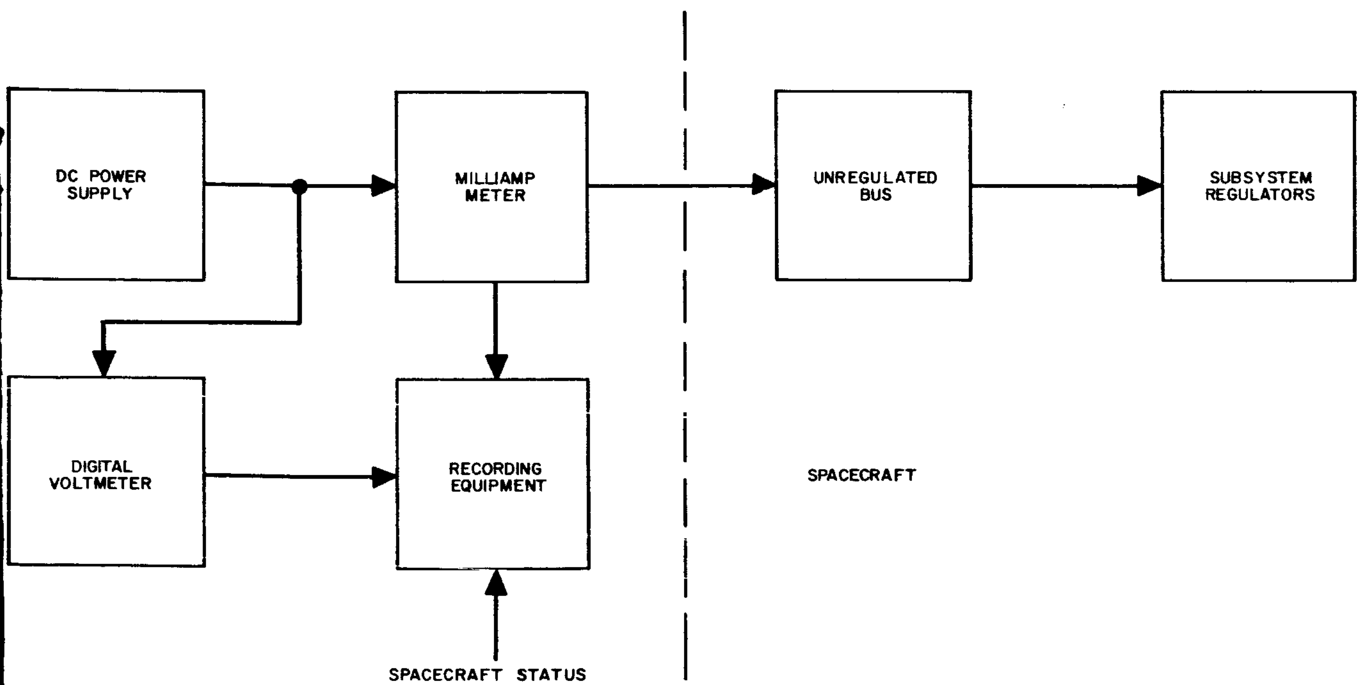
An external power supply will be adjusted to -26 volts, -32 volts, and -36 volts with a digital voltmeter. As each subsystem is commanded on, the current drawn by the spacecraft will be monitored by a milliamp meter and digitally recorded.

Equipment Required

1) Variable dc voltage power supply
2) Digital voltmeter
3) Milliamp meter
4) Command signal generator

Spacecraft Access Required

1) Unregulated bus



BATTERY CAPACITY TEST - TEST 36

Purpose of Test

To determine the capacity of the spacecraft batteries.

Source of Test

NASA Specification S2-0100, paragraph 3.8.31

Test Procedure

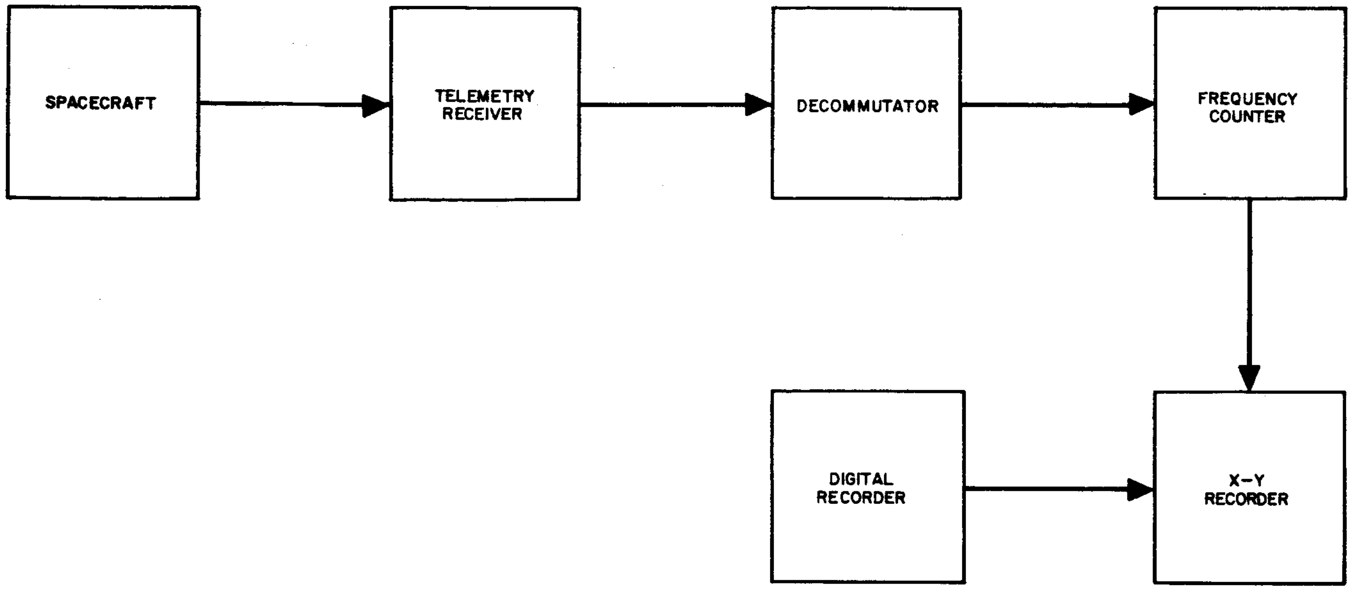
A telemetry transmitter and encoder will be turned on and operated by the spacecraft batteries for a predetermined time interval. The telemetered battery voltage will be plotted as a function of time on an x-y recorder. The batteries will be operated until the curve passes the knee of the voltage drop-off point.

Equipment Required

1) Command generator
2) Telemetry receiver and decommutator
3) Digital clock
4) Frequency counter
5) X-Y plotter

Space Access Required

None



System Block Diagram

The system block diagram (Figure 8-18) indicates the planned test equipment designed to meet the stated requirements. Semi-automation is obtained through the use of various patch panels and program boards. This philosophy allows a preassembled program board to be plugged into a patch panel for each series of tests, automatically connecting the proper equipment as outlined in the various test diagrams. Maximum versatility is retained through this use of patch panels, while intermodule action and pickup is held to a minimum by the use of four frequency separate panels. The equipment in this configuration is suitable for van mounting and capable of being used as a Mark II field test station.

The diagram indicates major equipment classification areas. RF signal generators and analysis equipment; recording devices providing both quick-look and delayed analysis records; video and audio frequency generation and analyses equipment; telemetry receiving, decommutating, command generation, transmission and synchronous controlling equipment; and specialized communication system test equipment. Four Hughes dual-mode transmitters and receivers permit modulation and detection of all carriers simultaneously, while special circuits listed under miscellaneous equipment handles tests on the remaining spacecraft systems (actual timer, sun sensors, etc.).

Master Index

The Index of Equipment given in Table 8-16 is compatible with the Block Diagram (Figure 8-21). It reflects the requirements of both functions of the equipment, telemetry and command, as well as System Tests.

Equipment Peculiar to T&C Ground Station

T&C Station Housing

The Hughes proposal for Syncom II development and launch program SSD 3127 dated 21 March 1963 includes a concept of air-inflated structures to house ground telemetry and control equipment. The desirability of this concept is predicated on superiority in the following aras:

1) Comparable initial cost

2) Easily convertible to permanent facility

3) No constraints on type of aircraft for transport

4) Separate shipment (apart from electronics) capability

5) Weight saving

Areas 1, 3, 4, and 5 have been investigated during the interim period, resulting in the following comparative cost study:

	Van (10'HX8WX32L) Similar to Syncom I	Air-Inflatable Structure (11'HX20'WX48'L)
Cost Bare	\$15150	\$20500
Concrete Substructure	---	1000
Total Initial Cost	\$15150	\$21500
Air Transportation	25000	531
(Electronics)	Included in above	10620
Total First Installation Cost	\$40150	\$32651

The bases on which this evaluation was made are as follows:

1) Costs of van shipment via MATS C124 to Joburg (actuals)

2) Cost of commercial air shipment to Joburg (estimated on Pan Am rates of \$1.77 per pound for cargo over 1100 pounds.)

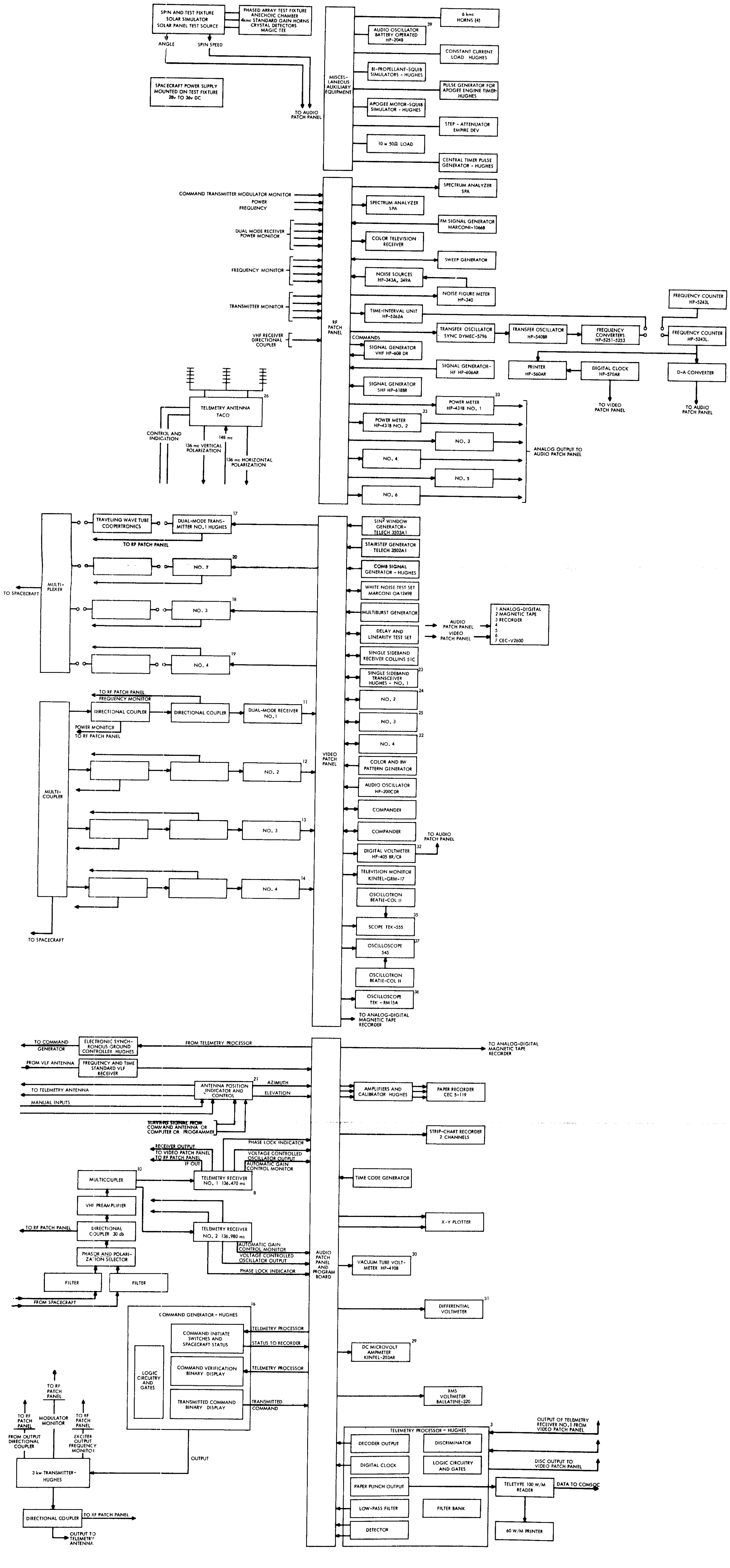


Figure 8-21. Syncom II Test Station

TABLE 8-16. INDEX OF EQUIPMENT

Quantity	Control Item Name	Source
1	Command Signal Generator	Hughes
1	Electronic Synchronous Ground Controller	Hughes
1	Telemetry Processor	Hughes
1	Telemetry and Command Simulator	Hughes
1	Audio Patch Panel	Hughes
1	Video Patch Panel	Hughes
1	RF Patch Panel	Trompeter Electronics
1	Telemetry Receiver No. 1	136.470 mc
1	Telemetry Receiver No. 2	136.980 mc
1	Telemetry Receiver Multi-coupler	Hughes
1	Dual-Mode Receiver No. 1	
1	Dual-Mode Receiver No. 2	
1	Dual-Mode Receiver No. 3	
1	Dual-Mode Receiver No. 4	
1	Multicoupler for Dual-Mode Receivers	
1	Telemetry Transmitter	Hughes
1	Dual-Mode Transmitter No. 1	Hughes
1	Dual-Mode Transmitter No. 2	Hughes

TABLE 8-16. (continued)

Quantity	Control Item Name	Source
1	Dual-Mode Transmitter No. 3	Hughes
1	Dual-Mode Transmitter No. 4	Hughes
1	Multiplexer for Dual-Mode Transmitters	
1	SSB Transceiver No. 1	Hughes
1	SSB Transceiver No. 2	Hughes
1	SSB Transceiver No. 3	Hughes
1	SSB Transceiver No. 4	Hughes
1	Telemetry and Command Antenna	
1	Telemetry Antenna Position Indicator and Controller	
1	RMS Voltmeter	Ballatine 320
1	DC Microammeter-Voltmeter	Kintel 203AR
1	VTUM	HP 410B
1	Differential Voltmeter	
1	Digital Voltmeter	HP 405CR
6	Power Meters	HP 431B
1	Phase Meter	AD-YU Electronics
1	Oscilloscope	Tek 555A
1	Oscilloscope	Tek 545A

TABLE 8-16. (continued)

Quantity	Control Item Name	Source
1	Oscilloscope	Tek RM15
2	Preamplifiers	Tek Type C-A, Dual Trace
1	Preamplifier	Tek Type H, Wideband, High Gain
1	Preamplifier	Tek Type L, Fast Rise High Gain
1	Preamplifier	Tek Type L Fast Rise High Gain
1	Preamplifier	Tek Type M, Four Trace
2	Scope Cameras	Beatle Coleman Mark II D
1	Mobile Score Cart	Tek 500/53A
1	Audio Oscillator	HP 200 ABR
1	Audio Oscillator	HP 204B
1	HF Signal Generator	HP 606AR
1	VHF Signal Generator	HP 608DR
1	SHF Signal Generator	HP 618BR
1	FM Signal Generator	Marconi 1066B
1	Stairstep Generator	Telech. 3502A1
1	Sin^2 Window Generator	Telech. 3503A1
1	Combination Signal Generator	
1	Sweep Generator	

TABLE 8-16. (Continued)

Quantity	Control Item Name	Source
1	Pulse Generator	
1	Magnetic Tape Recorder	
1	Recording Oscillograph	CEC 5-119
1	Strip Chart Recorder (two channels)	
1	Recorder Calibration Unit	
1	X-Y Plotter	
1	TTY 100 W/M Reader	
1	TTY 60 W/M Printer	
1	Paper Tape Punch	CY-2542
1	Digital Recorder	HP 560AR
1	Digital Printer	HP 565A
1	White Noise Test Set	Marconi OP12498
1	Noise Figure Meter	HP 340BR
1	Noise Source	
1	Color, Black and White TV Pattern Generator	
1	TV Monitor	Kintel GRM-17
1	Color TV Receiver	
1	Frequency and Time Standard ULF Receiver	
1	Digital Clock	HP 570AR
1	Time Code Generator	E. E. Corp.

TABLE 8-16. (Continued)

Quantity	Control Item Name	Source
2	Spectrum Analyzers	
1	TWT Amplifier	
1	Delay and Linearity Test Set	
1	Step Attenuator	Empire Devices
1	Transfer Oscillator Sync	Dymec 5796
1	Transfer Oscillator	HP 540BR
1	Frequency Converter	HP 5251
1	Frequency Converter	HP 5253A
1	Frequency Counter	HP 5243L
1	Time Interval Unit	HP 5262A
1	Digital Analog Converter	HP 580A
1	Spin and Test Fixture	Hughes
1	Phased Array Test Fixture	
1	Spacecraft Power Supply	Hughes
1	Solar Light Simulator	Hughes
1	Solar Panel Test Light Fixture	Hughes
1	Pulse Generator for Apogee Engine Timer	Hughes
1	Bi-Propellant Squib Simulator	Hughes
1	Electronic PSI Simulator	
1	Apogee Motor Squib Simulator	Hughes

TABLE 8-16. (Continued)

Quantity	Control Item Name	Source
1	Ten Watt Fifty Ohm Load	Hughes
1	Test Programmer	
1	SSB Receiver	Collins Mod No. 51C

3) Differences in cost of air-conditioning van and air-inflatable structure assumed negligible and not included.

4) Differences in cost of light and power installations for the two alternates assumed negligible and not included.

5) Cost of air-inflatable structure based on BirdAir proposal P-72 (see appendix)

These ground rules are considered to be quite realistic based on recent experience. Considerations of alternative destinations and modes of transportation change the picture on a proportionate basis with equal first cost occurring at an approximate shipping radius of 2500 miles. Even sea transportation (which has never been possible for foreign stations) favors the air-inflatable structure since its volume and weight are only 15 percent and 40 percent respectively that of the van and equipment.

Other comparative analyses can be made on the basis of cost per unit, floor space, and cost of resiting; using the Johannesburg figures, the van costs are $190 per square foot while the air-inflatable housing costs are $50 per square foot. These figures are derived from a useable 210 square foot or 7-1/2 x 28 in the van and 638 square foot or 14-1/2 x 44 in the air-inflated structure.

The differences in siting costs are estimated at $1700, broken down as follows:

Concrete substructure unrecovered	$1000
Power, wiring, and lighting unrecovered	$ 300
Time loss in assembly and disconnection	$ 400

Conclusions:

1) Initial acquisition costs are approximately 30 percent higher for the air-inflatable structure.

2) Transportation costs are approximately 50 percent lower for the air-inflatable structure. Reduced cube, weight, and size account for this significant reduction.

3) Electronic equipment can be shipped in two rack bays and will be compatible with any commercial aircraft including Boeing 707.

4) Resiting differences are nominal.

5) The larger, more convenient, operating area afforded by the air-inflatable structure at realizable savings makes this housing most desirable.

Description of Air-Inflatable Housing

Size:

Overall length	48 feet 7-1/2 inches
Inside length	45 feet 3-1/2 inches
Overall width	20 feet 2 inches
Inside width (at floor)	16 feet 10 inches (approximately)
Overall height	11 feet 8 inches
Inside height (center)	10 feet 0 inches

Material:

Translucent white vinyl coated nylon fabric having an overall weight of 22-23 ounces per square yard. Base fabric weight 5.5 ounces per square yard.

Construction:

All primary load-carrying joints electronically welded.

Anchorage:

Equipped for anchorage to a concrete slab. Preinstalled concrete anchorage by customer. All other anchor hardware by supplier. Vertical tension rating of concrete anchors to be not less than 4000 pounds.

Access:

34 inches by 6 foot 2 inches personnel door, equipped with 3 inches thick base platform, door closure, and outdoor hardware (including lock) will be provided at each end of the structure.

An equipment access opening measuring 7 feet by 7 feet will be provided at one end of the structure (by deflating the end dual-wall section, disconnecting from the end single curtain, and rolling up). Cable, tube, and waveguide openings will be provided as follows:

One 8-inch opening, floor level, one end

Two 8-inch openings, floor line, center one side

Each opening would be furnished with a sleeve and suitable straps for closing around the cable or tubes.

Windows:

One clear lucite window will be provided in each end, measuring approximately 2 feet by 3 feet.

Pressurization:

A dual blower, fully automatic system would be provided. Blower operation would be cyclic, controlled by pressure switches. Blower motors would be powered by 208-volt, three-phase, 60-cycle motors. Blowers would be packaged in a weather protective housing for installation outdoors, adjacent to the side of the structure. All ducting, manifolding and valving would be furnished. The customer would furnish line power hookup and a fully automatic emergency power source. It is anticipated that each blower would be powered by a 1-1/2 horsepower motor.

Ventilation:

Although the structure will be heated and air-conditioned separately, modest ventilation will be provided by a small two-speed fan located at the center of one side. This fan would discharge into the structure; outflow would be through the four exit vents (adjustable openings).

Heat Transfer Characteristics:

The overall coefficient for the structure is 0.6 Btu/°F/square feet per hour with an average wind of 10 to 15 mph. The midsummer solar heat load would be approximately 16,500 Btu/hour. The single end curtains would be foam-insulated.

Floor:

A simple, low cost, expendable floor is best constructed at the site. It is suggested that the construction consist of a polyethylene diaphragm at the concrete, 1/2 inch exterior plywood, 2 by 4 joists (on side) on 12 inch centers, 3/4 inch exterior plywood, felt paper and tile, or linoleum. A seal flap at the bottom of all cells to cleat to the edge of the floor is provided with the housing.

Installation Service:

Hughes would furnish handling labor and electrical hookup. Excluding the electrician, six laborers should be adequate to install the structure in less than one day (assuming that all anchors, floor, and advance site preparation is completed in advance).

Miscellaneous:

A fabric repair kit, and an Installation, Maintenance, and Repair Manual (six copies) would be supplied by vendor.

HANDLING EQUIPMENT

Hoisting Sling

A combination spacecraft and apogee motor sling was designed and fabricated (Figure 8-22). The sling is capable of lifting the spacecraft from either end by four attach points, and the addition of simple adapters (Figure 8-23) converts it to an apogee motor hoisting sling to ensure that no sudden shock loads, during lifting, are transmitted into the spacecraft structure or apogee motor casing, the four cables are shock mounted. The cable shock absorbers also function as load equalizers.

Mobile Assembly Fixture

Various assembly fixture concepts are being investigated. The primary function of the fixture is to support the spacecraft basic structure during assembly and maintenance operations.

Figures 8-24 and 8-25 illustrate two concepts under investigation. In Figure 8-24, the fixture incorporates a cantilevered arm on which the spacecraft structure is mounted. The arm can be rotated about a horizontal axis by means of a gear reductor. In Figure 8-25, the spacecraft can be rotated about the vertical axis. The adapter plate is ball-joint-mounted, which allows the spacecraft to be rotated at various angles.

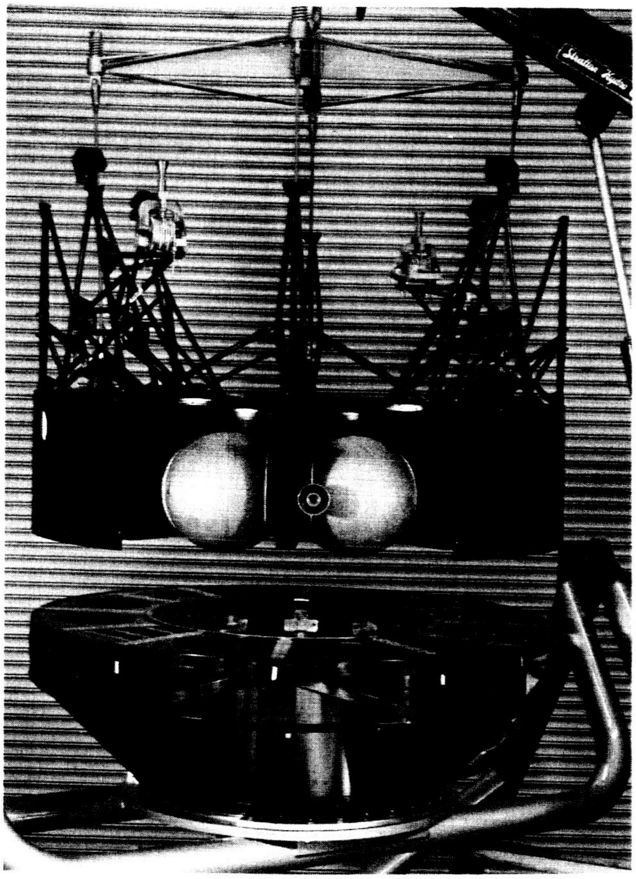


Figure 8-22. Combination Spacecraft and Apogee Motor Sling

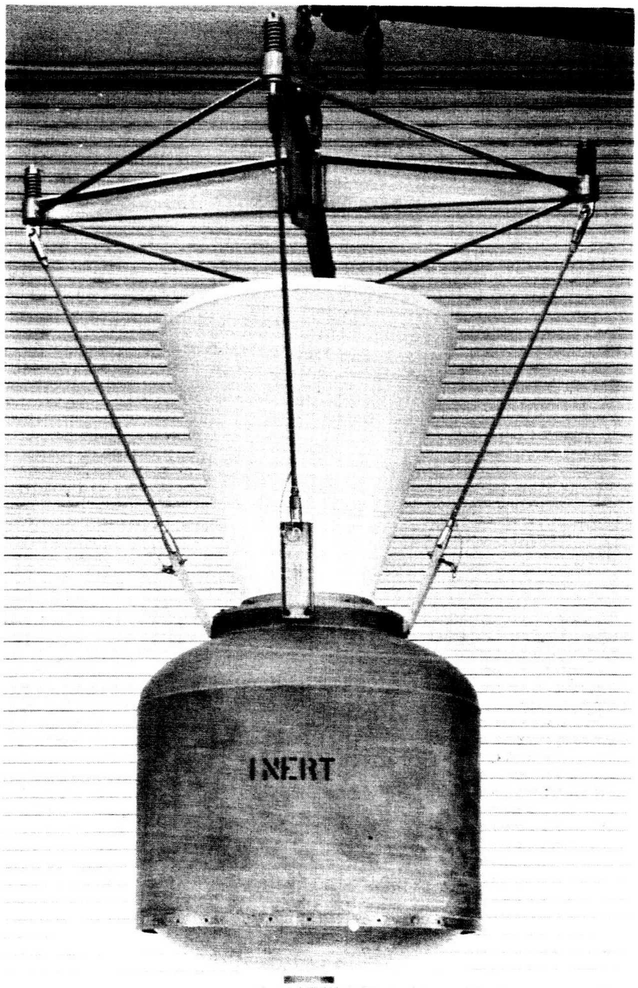


Figure 8-23. Apogee Motor Hoisting Sling

Figure 8-24. Assembly Fixture with Cantilevered Arm

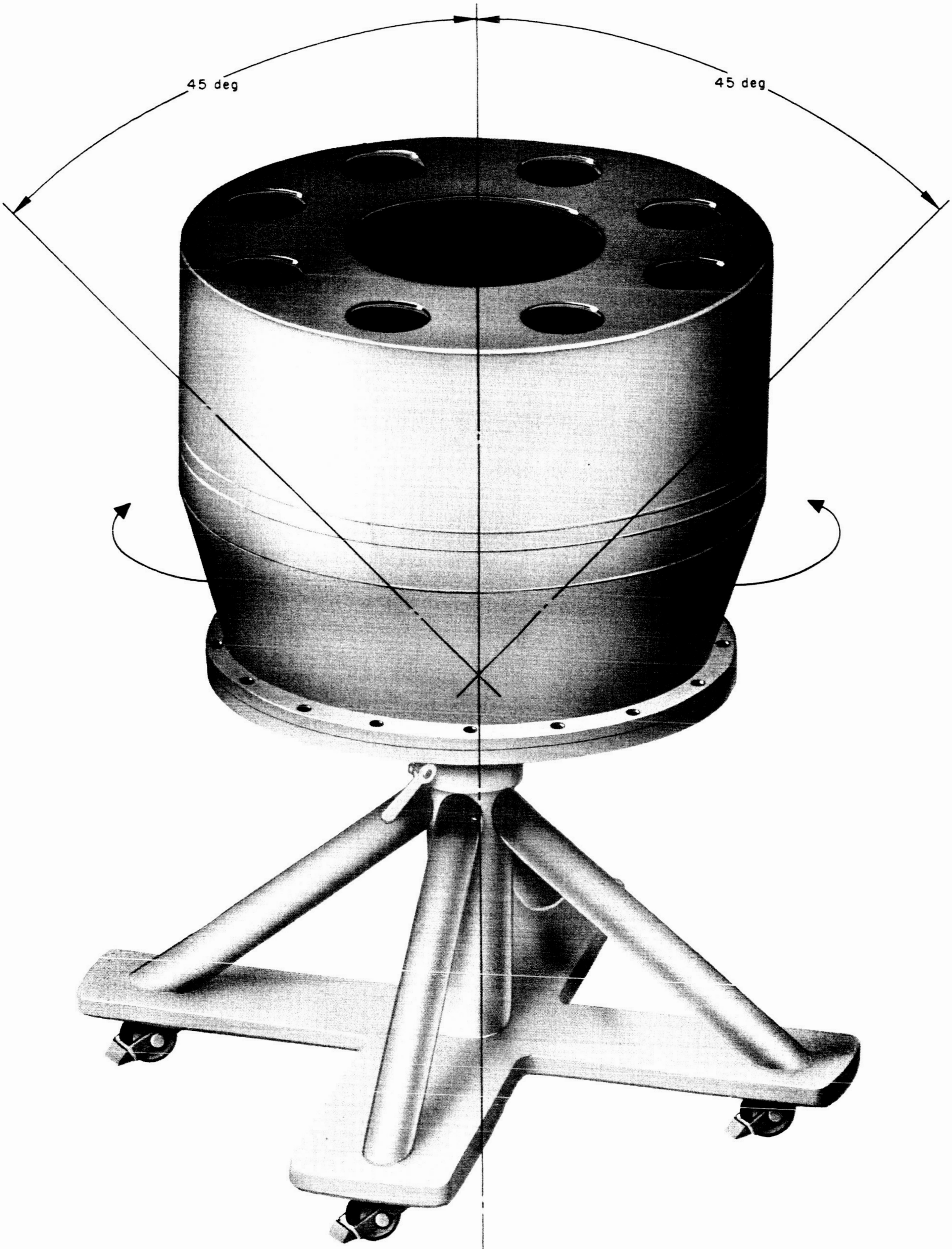


Figure 8-25. Assembly Fixture with Gear Reducer

9. QUALITY CONTROL OPERATING PLAN

INTRODUCTION

The preliminary Quality Control Operating Plan defined herein is applicable to all Syncom II operations at the Culver City and El Segundo plants. The program is prepared in accordance with Aerospace Group policies, Space Systems Division policies, Syncom Project requirements, and applicable portions of NASA Quality Publication NPC 200-2. The program plan is authorized by El Segundo Quality Assurance Bulletins and Culver City Quality Control Administrative Procedures issued by the respective Quality Managers in accordance with the responsibility placed upon them by the Vice President-Manager of the El Segundo Division and the Director of Product Effectiveness, Aerospace Group.

The Quality Control Instructions (QCI) referenced herein have been prepared primarily to govern the Culver City Syncom Quality Control operations. These instructions have been adapted for use at the El Segundo facility during the interim required to prepare El Segundo Departmental Instructions. As the El Segundo Instructions are prepared, they will be incorporated into the program plan and will supersede the QCI for the El Segundo operation.

DESCRIPTION OF PROJECT

The Syncom II program includes the development, fabrication, test, and mission operations of spacecraft to be orbited by Atlas-Agena D launch vehicles and operated in conjunction with NASA communication ground stations for the performance of station-orbit experiments.

The planned program includes development of an advanced engineering model spacecraft for systems integration and test. This system will be used to finalize the design of the various subsystems and will therefore conform as nearly as possible to flight hardware form factors and weights. Fabrication of equipment for this system may be accomplished under informal drawings and no formal burn-in of components will be required.

Prototype and flight spacecraft will be fabricated to formal, released drawings under NASA quality standards. The only planned difference between the prototype (qualification test) spacecraft and the flight spacecraft is the extent of the power aging of components prior to selection for fabrication. The components for the prototype spacecraft will receive burn-in and selection under criteria similar to the components used in the Syncom I flight spacecraft. The components for Syncom II flight spacecraft systems will receive extensive power aging prior to selection for fabrication.

The program will require communication transponders packaged in convenient carrying cases and telemetry and command (T & C) simulators for use in integration and prelaunch checkout testing of the ground communications and T & C stations.

The spacecraft systems have been functionally divided into major control items and minor control items. Test sets or setups are required for checkout and test of assembled major control items prior to acceptance for assembly into spacecraft.

System test equipment will be developed to permit operation of the various systems and conduct of frequent performance checks of system parameters. In addition, the system test equipment will provide the basic means for initial integration and checkout of the assembled spacecraft and subsequent occasional detailed examinations of the internal system parameters.

QUALITY REQUIREMENTS

General

The quality requirements for the Syncom II Development and Launch Program follow the general requirements of the NASA Quality Publication NPC 200-2, "Quality Program Provisions for Space Systems Contractors." The definitive contract and Syncom Project directives will provide singular task requirements not specifically identified by NPC 200-2.

Revisions

The following paragraphs of NPC 200-2 will be modified as shown:

2.2 Quality Program Documentation (Appendix B) - The Contractor shall submit the following documents for approval by the NASA:

a. Qualification Status List

b. Major Control Item Test Plan

The contractor shall submit the following for review:

a. Quality Control Operating Plan

b. Storage Procedure for End Items

The contractor shall submit the following for information:

a. Monthly Quality Status Report

b. Quarterly Summaries of Quality Program Performance Audits

The contractor shall make the following available for review upon NASA request at the Contractor's facility:

a. Test and Inspection Procedures

b. Process Control Procedures

c. Results of Special Measuring and Test Equipment Evaluation

d. Special Sampling Plans

4.4 Identification

Articles mutually designated by the Project Managers of Hughes and the NASA shall be identified by a unique number and, if required, a serial number. Like items in large quantities, such as resistors and capacitors, shall be identified by lot number.

5.3.1.d Subcontractor Quality Program - Supplier quality program requirements will be predicated on the specific part being purchased.

5.3.1.g Evidence of Supplier Inspection Performed - Evidence of supplier inspections shall be in the form of a completed inspection status tag or a certificate of compliance. Records need not be submitted, but must be made available upon contractor or customer request.

7.3.1 Inspection and Test Procedures - Detailed written procedures will be prepared for specified processes, operations, and items.

8.4 Rework without MRB - Disposition of nonconforming material in the categories of "return for completion of operations" or "return for rework to drawing" may be authorized by Quality Control without formal material review.

13.1 Training - The contractor shall develop, implement, and maintain training programs for quality control and manufacturing personnel who may have an effect upon quality.

13.2 Certification of Fabrication and Inspection Personnel - Contractor personnel responsible for performing special fabrication and inspection operations of a specialized nature having a significant effect upon quality shall be certified. These operations presently include welding, micro-resistance welding, soldering, wiring, dye penetrant, X-ray, radiography, and particle detection.

ASSIGNMENT OF QUALITY CONTROL LEVELS

Quality Control Levels are assigned in accordance with Quality Control Procedure 5-2.2 "Hardware Models, Type Designations and Quality Control Requirement." The levels are assigned to Syncom II hardware as follows:

Item	Level
T-1 ATD Engineering Model	I
T-2 Advanced Engineering Model	I
Y-1 Qualification Test Model (Prototype)	III
Y-2 Control Item Qualification Test Hardware	III
Flight Spares	
F-1, F-2, F-3, F-4	III
NASA Transponders	II
T & C Simulators	II
Test Equipment	II
Ground Control Equipment	III
Trailer Modification	II

Item	Level
Ground Handling Equipment	II
Test Fixtures	II
Handling and Shipping Containers	II

DESCRIPTION OF ORGANIZATION FOR IMPLEMENTATION OF THE QUALITY CONTROL OPERATING PLAN

The descriptions will be supplied at a later date on the following:

Quality organization

Flow of authority

QUALITY CONTROL INSTRUCTIONS

All general Quality Control Instructions (QCIs) will apply unless otherwise specified. Special project oriented QCIs and Quality Control Bulletins (QCBs) will be developed and issued as required. The general QCIs are listed in categories as they apply to NPC 200-2. Engineering Procedures (EPs) and tentative project QCIs are listed where they are applicable.

5.1 Basic Requirements

5.1.1 Documentation -

QCI A	"Quality Control Instruction, Initiation and Preparation"
QCI 1.1-3	"Quality Control Reporting - All Fabrication Areas"
QCI 1.1-1	"Use of HAC Form 380A"
QCI 1.1-2	"Use of HAC Form 1353 A - Inspection Status"
QCI 1.1-12	"HAC Form 3236, Variance Authorization"
QCI 1.1-13	"HAC Form 672A, Substitution Authorization"
QCI 2.2-6	"Quality Control Reporting Receiving Inspection"
QCI 1.1-17	"Use of HAC Form 359, Inspection and Test Report"

5.2 Management

5.2.1 Planning -

QCI 4.1-1	"Quality Control Operating Plan"

5.3 Design and Development Control

EP 4-9	"Engineering Change Management - Development"
EP 7-17-1	"Advance Syncom Project - Engineering Data"
EP 9-1	"Design Review"

5.4 Control of Contractor Procured Material

5.4.1 Selection of Procurement Sources

QCI 1.1-9	"Quality Capability Surveys of Suppliers"
QCI 1.1-14	"Pre-Procurement Source Selection - Major Subcontract"
QCI 2.2-16	"Evaluation of Special Process Suppliers"

5.4.2 Procurement Documents Review

QCI 2.2-15	"Quality Control Screening - Procurement Documents"
QCI 3.13-	"Syncom II - Screening Purchase Orders for Subcontract Items"

5.4.3 Government Source Inspection

QCB A	"Air Force Policy Item List"

5.4.4 Contractor Source Inspection

QCI 2.1-10	"HAC Source Inspection - Inspection and Test Section"
QCI 2.2-9	"Source Inspection - Supplier Evaluation Section"

5.4.5 Receiving Inspection

QCI 2.2-1	"Receiving Inspection Sampling Plan"
QCI 2.2-2	"Receiving Inspection Instruction Sheet"
QCI 2.2-3	"Raw Stock Receiving Inspection"
QCI 2.2-6	"Quality Control Reporting - Receiving Inspection"
QCI 2.2-7	"Receiving Inspection - General Requirements and Responsibilities"
QCI 2.2-10	"Materials and Processes Acceptance Requirements (MAPAR) Testing"
QCI 2.2-12	"High Reliability Receiving Inspection"

5.4.6 Supplier Rating and Preferred Source List

QCI 2.2-4	"Vendor Quality Rating System"

5.5 Control of Government-Furnished Property (GFP)

5.5.1 Inspection of GFP

QCI 2.2-14 "Receiving Inspection of GFP and Bailed Property"

5.5.2 Defective GFP

QCI 2.2-13 "Control of Damaged GFP - Receiving Inspection"

5.6 Control of Contractor-Fabricated Articles

5.6.1 General

QCI 3.13- "Syncom II - Screening Manufacturing Requests for Quality Control Requirements"
QCI 3.13- "Syncom II - Screening Work Orders for Quality Control Requirements"
QCI 3.13- "Syncom II - Screening Assist Work Authorizations"

5.6.2 Inspection and Test Planning

QCI 4.1-1 "Quality Control Operating Plan"
QCI A "Quality Control Instructions Initiation and Preparation"

5.6.3 Inspection and Test Performance

QCI C "Inspection and Test - General Requirements and Responsibilities"
QCI 4.1-7 "Test Verification"
QCI 2.1-9 "Final Inspection of Missiles and Space Vehicles"
QCI 3.13- "Syncom II Spacecraft Final Inspection and Shipping Requirements"

5.6.4 Fabrication Controls

QCI 2.1-5 "Inspection of Micro-Resistance Welding"
QCI 4.1-3 "Certification of Metal Finishing Process"
QCI 4.1-4 "Qualification/Certification of Heat Treating"
QCI 4.1-8 "Qualification of Micro-Resistance Welding Machines"
QCI 4.1-9 "Welding Inspection, Fusion"
QCI 4.1-10 "Heat Treating Inspection"
QCI 4.1-11 "Plating Inspection"
QCI 4.1-12 "Painting Inspection"

5.7 Nonconforming Material

QCI 1.1-5 "Control of Nonconforming Supplies"
QCI 2.2-11 "Quality Control Screening of Scrap or Salvageable Material"

5.8 Inspection, Measuring, and Test Equipment

QCI 1.1-10 "Measurement and Test Equipment Surveys"
QCI 2.1-8 "Tooling Inspection"
QCI 4.1-14 "Accuracies of Test Measurement Equipment"

5.9 Inspection Stamps

QCI 1.1-2 "Use of HAC Form 1353A - Quality Control Inspection Status"

5.10 Preservation, Packaging, Handling, Storage, and Shipping

QCI 2.1-6 "Periodic Inspection of Stores"
QCI 2.2-8 "Packaging and Shipping Inspection"
QCI 3.13- "Syncom II - Bonded Stores Inspection"
QCI 3.13- "Syncom II Spacecraft Final Inspection and Shipping Requirements"

5.11 Statistical Quality Control

QCI 2.2-1 "Receiving Inspection Sampling Plan"

5.12 Training and Certification of Personnel

QCI 4.1-2 "Certification of Micro Resistance Welding Operators"
QCI 4.1-5 "Certification/Qualification of Class "A" Welding Operators"
QCI 4.1-13 "Welding, Resistance, Inspection Qualification"

5.13 Data Reporting and Corrective Action

QCI 1.1-7 "Control Point, Corrective Action Requests"
QCI 1.1-8 "Use of HAC Form 805, Corrective Action Request"
QCI 1.1-3 "Quality Control Reporting - All Fabrications Areas"

5.14 Audits

QCI 1.1-15 "Internal Quality Audits"

Reference

1. MIL-Q-9858.
2. MIL-Q-21549 (NOrd).
3. MIL-STD-243.
4. MIL-I-45208 (ORD).
5. NASA-NPC 200-2.
6. NASA-NPC 200-3.
7. Other specifications as contractually required.

Applicable to

1. Aeronautical Systems - Division 21
2. Space Systems - Division 22
3. Research and Development - Division 27
4. Guidance and Controls -Division 29

General

To provide "Standard Model Designations" for the various types of hardware and to establish "Quality Levels" based upon the quality control requirements for each model.

Model Designations

For the purpose of establishing uniform definitions for the vari-our types of hardware produced, the following model designations from MIL-STD-243 are described in Attachment "A" (page 6).

1. Breadboard Model.
2. Experimental Model.
3. Developmental Model.
4. Service Test Model.
5. Prototype (Preproduction) Model.
6. Production Model.

Quality Control Levels

Contractual requirements, phase of procurement and type of hardware to be produced dictate the level of quality required. This procedure establishes three quality control levels to provide the latitude necessary to product the various model types. Model designations are therefore grouped under the levels shown in Table I.

Certified: E. J. Burns Date: December 14, 1962

Determination of Quality Control Levels

Concurrent with the input to a Request for Proposal/Quotation, Quality Control determines the required level of inspection. Variations from normally specified levels may be indicated by the contract. Such variations are called out in the Quality Control Operating Program Plan. Concurrence in the assignment of quality control levels are obtained from the Project Manager.

Assignment of Multi-Levels

More than one level may be required for a specific contract. In such cases, the Quality Control Operating Plan is clearly defined and identifies the hardware or the periods through which the various levels shall apply.

Applicable Quality Control Procedures

Table I contains a list of Quality Control procedures by title and procedure number. Each column (Levels I, II and III) bears a series of notations as to the applicability of a specific procedure to the quality level under consideration.

Relationship Between Quality Levels and Model Types

Attachment "B" (pages 7 through 10) establishes the relationship between the quality control levels in Table I and the model designations of MIL-STD-243. It summarizes the level of effort required for significant elements of the Quality Control Operating Plan and is intended for distribution to other Aerospace Group organizations to assist in anticipating program costs and schedules.

TABLE I
APPLICABILITY OF QUALITY CONTROL PROCEDURES TO QUALITY LEVELS

PROCEDURE NUMBER	PROCEDURE TITLE	PROCEDURE APPLICABILITY Q. C. LEVEL I	Q.C. LEVEL II	Q.C. LEVEL III
5-1.0	POLICY, OBJECTIVES, AND ORGANIZATION			
5-1.1	Policy and Objectives	Applies	Applies	Applies
5-1.2	Organization and Functions	Applies	Applies	Applies
5-1.3	Q.C. Procedures, Instructions & Bulletins	Applies as noted	Applies as noted	Applies as noted
5-2.0	CONTRACTUAL REQUIREMENTS			
5-2.1	Contractual Quality Requirements	Applies	Applies	Applies
5-2.2	Hardware Models, Type Designations and Quality Requirements	Applies	Applies	Applies
5-2.3	Quality Control Operating Program Plan	As required by contract	As required to clarify contract requirements	Applies
5-2.4	Proposal Preparation	Applies	Applies	Applies
5-3.0	INSPECTION AND TEST			
5-3.1	Workmanship Standards	Applies as required by contract & specified in the Q.C. Program Plan	Applies	Applies
5-3.2	Quality Control Instructions	Applies	Applies	Applies
5-3.3	In-Process Inspection and Test	As required by contract	Applies	Applies
5-3.4	Special Processes Control	Applies	Applies	Applies
5-3.5	Inspection Stamps & Indication of Inspection Status	As required by contract	Applies	Applies
		As required by contract	Applies	Applies
5-3.6	Test Verification	As required by contract	Applies	Applies
5-3.7	Final Acceptance	As required by contract	Applies	Applies
5-3.8	Shipping Inspection	Applies	Applies	Applies

TABLE I
APPLICABILITY OF QUALITY CONTROL PROCEDURES TO QUALITY LEVELS

PROCEDURE NUMBER	PROCEDURE TITLE	PROCEDURE APPLICABILITY		
		Q.C. LEVEL I	Q.C. LEVEL II	Q.C. LEVEL III
5-4.0	CONTROL OF PURCHASED MATERIAL & PARTS			
5-4.1	Procurement Source Selection	Applies	Applies	Applies
5-4.2	Procurement Document Control 1. Purchase Orders & Purchase Req's. 2. Assist Work Authorizations 3. Purchase Order Quality Attachments	Applies	Applies	Applies
5-4.3	Subcontractor & Supplier Control	As required by contract	Applies	Applies
5-4.4	Source Inspection	As required by Engineer- and Quality Control	As required by Engi- neering & Q.C.	Applies
5-4.5	Government Source Inspection	As required by the ap- propriate Govt. Agency	As required by the propriate Govt.agency	As required by the approp.Govt.agency
5-4.6	Receiving Inspection and Test	Applies	Applies	Applies
5-4.7	Supplier Evaluation	Applies	Applies	Applies
5-4.8	Certification of Supplier Special Processes and Personnel	Applies	Applies	Applies
5-4.9	Control of Stores	Applies	Applies	Applies
5-5.0	CONFIGURATION CONTROL			
5-5.1	Product Drawing & Change Control	Applies	Applies	Applies
5-5.2	Design Review	Applies	Applies	Applies
5-6.0	CONTROL OF NONCONFORMING SUPPLIES			
5-6.1	Control of Nonconforming Material	Applies	Applies	Applies
5-7.0	GOVERNMENT PROPERTY CONTROL			
5-7.1	Control of Government Furnished and Bailed Property	Applies	Applies	Applies
5-7.2	Control of Facilities Equipment	Applies	Applies	Applies
5-8.0	STATISTICAL QUALITY CONTROL			
5-8.1	Statistical Methods and Controls	Applies	Applies	Applies

TABLE I
APPLICABILITY OF QUALITY CONTROL PROCEDURES TO QUALITY LEVELS

PROCEDURE NUMBER	PROCEDURE TITLE	PROCEDURE APPLICABILITY		
		Q.C. LEVEL I	Q.C. LEVEL II	Q.C. LEVEL III
5-9.0	DATA, REPORTING & CORRECTIVE ACTION			
5-9.1	Inspection Records, Reports, Forms & Files	Applies	Applies	Applies
5-9.2	Corrective Action	Applies	Applies	Applies
5-10.0	QUALITY AUDIT			
5-10.1	Internal Quality Audits	Applies	Applies	Applies
5-11.0	CONTROL OF INSPECTION & TEST EQUIPMENT			
5-11.1	Preliminary Evaluation of Inspection and and Test Equipment	Applies	Applies	Applies
5-11.2	Calibration and Maintenance of Inspection and Test Equipment	Applies	Applies	Applies
5-12.0	TRAINING			
5-12.1	Indoctrination and Training of Quality Control Personnel	Applies	Applies	Applies
5-13.0	GOVERNMENT INSTALLATIONS AND TEST SITES			
5-13.1	Installation and Test Site Quality Control	Applies	Applies	Applies

Attachment "A"

MODEL DESIGNATIONS

Breadboard Model	An assembly of preliminary circuits and parts to prove the feasibility of a device, circuit, equipment, system, or principle in rough or breadboard form without regard to the eventual over-all design or form of parts.
Experimental Model	A model of the complete equipment to demonstrate the technical soundness of the basic idea. This model need not have the required final form or necessarily contain parts of final design.
Developmental Model	A model designed to meet performance requirements of the specification'or establish technical requirements for production equipment. This model need not have the final form or necessarily contain parts of final design. It may be used to demonstrate the reproducibility of the equipment.
Service Test Model	A model to be used for test under service conditions for evaluation of suitability and performance. It shall closely approximate the final design, have the required form, and employ approved parts or their interchangeable equivalents.
Prototype (Preproduction) Model	A model suitable for complete evaluation of mechanical and electrical form, design, and performance. It shall be of final mechanical and electrical form, employ approved parts and be completely representative of final equipment.
Production Model	A model in its final mechanical and electrical form of final production design made by production tools, jigs, fixtures, and methods.

QUALITY CONTROL LEVEL I
BREADBOARD AND EXPERIMENTAL MODELS

QUALITY REQUIREMENTS	Q.C. PROC. NO.	APPLICATION	
		BREADBOARD	EXPERIMENTAL
INSPECTION AND TEST	5-3.3	MECHANICAL - Inspect to information available, including marked changes. Mechanical configuration limited only by equipment and personnel safety considerations.	Same except layout drawings and sketches may be available. Mechanical configuration to meet planned tests and safety considerations.
	5-3.3	ELECTRICAL - Inspect for cold solder, possible shorts, discontinuities, damage, etc. Check continuity to preliminary schematics.	Same except check workmanship of electrical fabrication. Inspect to engineering information.
	5-3.3	TEST - Conduct test to instructions provided by Engineering. No witness required on tests by Engineering.	Same
	5-3.6	PROCESS - Assure compliance with instructions provided by Engineering	Assure compliance to design requirements
ENGINEERING DATA	5-5.1	Preliminary engineering sketches, drawings or instructions. No change control required	Same
WORKMANSHIP	5-3.1	To the extent required for mechanical and circuit integrity and for safety. Electrical connections require solder or pressure type connectors. Neatness and uniformity are desirable.	Same except HAC standards for component and wiring spacing should be observed. Marking and mounting hardware subject only to neatness and uniformity.
PARTS AND MATERIALS	5-4.0	May be provided from "Open" or "Controlled" stores. Part and component application should be evaluated and compared with specified limits for parts. HAC Preferred Parts to be given preference, considering circuit requirements and product improvement.	Same

Attachment "B"
(continued)

QUALITY CONTROL LEVEL I

BREADBOARD AND EXPERIMENTAL MODELS

QUALITY REQUIREMENTS	Q.C. PROC. NO.	APPLICATION	
		BREADBOARD	EXPERIMENTAL
CALIBRATION OF INSPECTION AND TEST EQUIPMENT	5-11.2	Standard test and gaging equipment to bear evidence of current calibration. Special breadboard equipment to have calibration capability.	Same
INSPECTION AND TEST RECORDS	5-9.1	Adequate to meet contract requirements	Same
NONCONFORMING PARTS AND MATERIAL	5-6.1	Applies to Receiving Inspection and Test to engineering data provided and on request of Engineering. Quality Control concurrence not required to accept nonconforming materials or parts.	Same
REMARKS		1. The above quality requirements pertain to deliverable items only. 2. For non-deliverable items, inspection and surveillance will be at Engineering request and limited to good "shop practice."	Same

QUALITY CONTROL LEVEL II
DEVELOPMENTAL MODELS

Attachment "B"
(continued)

QUALITY REQUIREMENTS	Q.C. PROC. NO.	APPLICATION
INSPECTION AND TEST	5-3.3	MECHANICAL - Inspection to available drawings. Marked drawings showing Engineering approval will be considered valid. Discrepancies to be recorded on applicable Inspection/Test records.
	5-3.3	ELECTRICAL - Inspection of assembly and circuitry to available schematics, wiring diagrams, and drawings. Discrepancies will be recorded on the applicable Inspection/Test records.
	5-3.3	TEST - Conduct or witness tests to available test specifications.
	5-3.6	PROCESS - Government, Hughes Process or Laboratory Services Process Instructions shall take precedence over preliminary specifications. Preliminary specifications are permissible in the event no Government, Hughes Process or Laboratory Services Process Instructions specifications are available.
ENGINEERING DATA	5-5.1	Latest available drawings, sketches, specifications, etc. Drawing change control in accordance with applicable procedures.
WORKMANSHIP	5-3.1	According to Shop Standards Manual.
PARTS AND MATERIALS	5-4.0	It is desirable that parts and materials be selected from MIL STD or HAC approved parts lists. The use of non-standard parts and materials should be limited to parts or materials which will in no way degrade the quality or the integrity of the assembly.
CALIBRATION OF INSPECTION AND TEST EQUIPMENT	5-11.2	Test and gaging equipment must bear evidence of current calibration. Records must be available.
INSPECTION AND TEST RECORDS	5-9.1	Adequate to meet contract requirements.
NONCONFORMING PARTS AND MATERIAL	5-6.1	Discrepant parts and materials to be suspended and dispositioned by Material Review.

QUALITY CONTROL LEVEL III
SERVICE TEST, PROTOTYPE (PREPRODUCTION) AND PRODUCTION MODELS

Attachment "B"
(continued)

QUALITY REQUIREMENTS	Q.C. PROC. NO.	APPLICATION: SERVICE TEST	APPLICATION: PROTOTYPE/PRODUCTION
INSPECTION AND TEST	5-3.3	MECHANICAL - Detailed inspection to standard released engineering drawings. First article acceptance of all tooling. Discrepancies to be recorded on applicable inspection records.	Same
	5-3.3	ELECTRICAL - Detailed inspection to be released engineering drawings, wiring diagrams, wire lists, and schematics. Discrepancies to be recorded on applicable inspection records.	Same
	5-3.3	TEST - To detailed test specifications.	Same
	5-3.6	PROCESS - Inspect to HP, LSPI or Government Specifications referenced on drawing.	Same
ENGINEERING DATA	5-5.1	Standard released engineering drawings and specifications, including formal change control.	Same
WORKMANSHIP	5-3.1	According to Shop Standards Manual or customers manual as applicable.	Same
PARTS AND MATERIALS	5-4.0	Controlled parts and materials must be selected from MIL-STD or HAC approved parts lists, unless specifically authorized by contract or contract waiver. HAC Preferred Parts to be given preference, considering leadtime, product improvement, and cost.	Same, except must be qualified to MIL-STD or HAC specifications.
CALIBRATION OF INSPECTION AND TEST EQUIPMENT	5-11.2	All test and gaging equipment to bear evidence of current calibration.	Same
INSPECTION AND TEST RECORDS	5-9.1	Records should show Inspection/Test history through final assembly, and maintained for period required by contract.	Same
NONCONFORMING PARTS AND MATERIAL	5-6.1	Discrepant materials and parts to be suspended and dispositioned by Material Review.	Same

6.0 Flow Charts

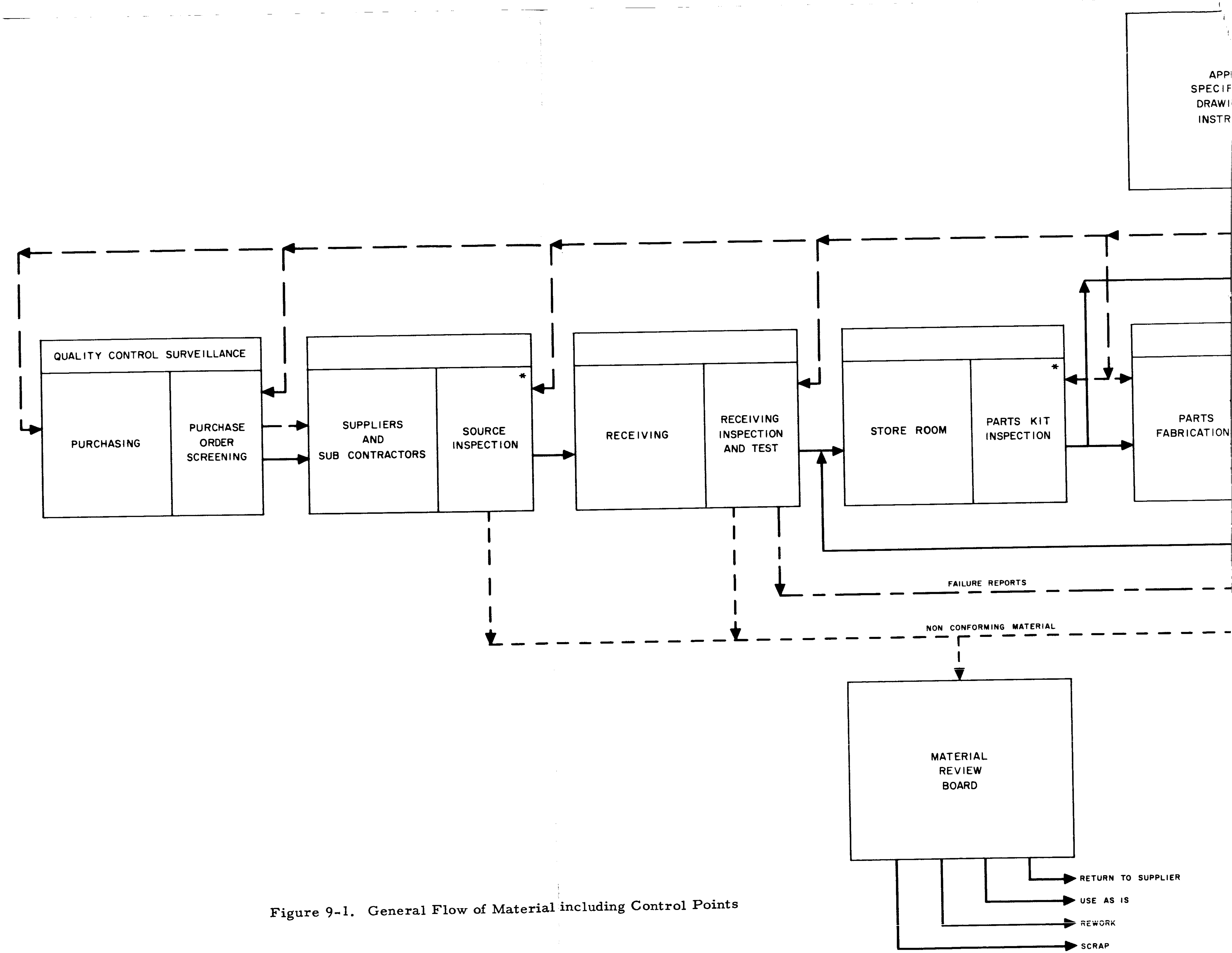
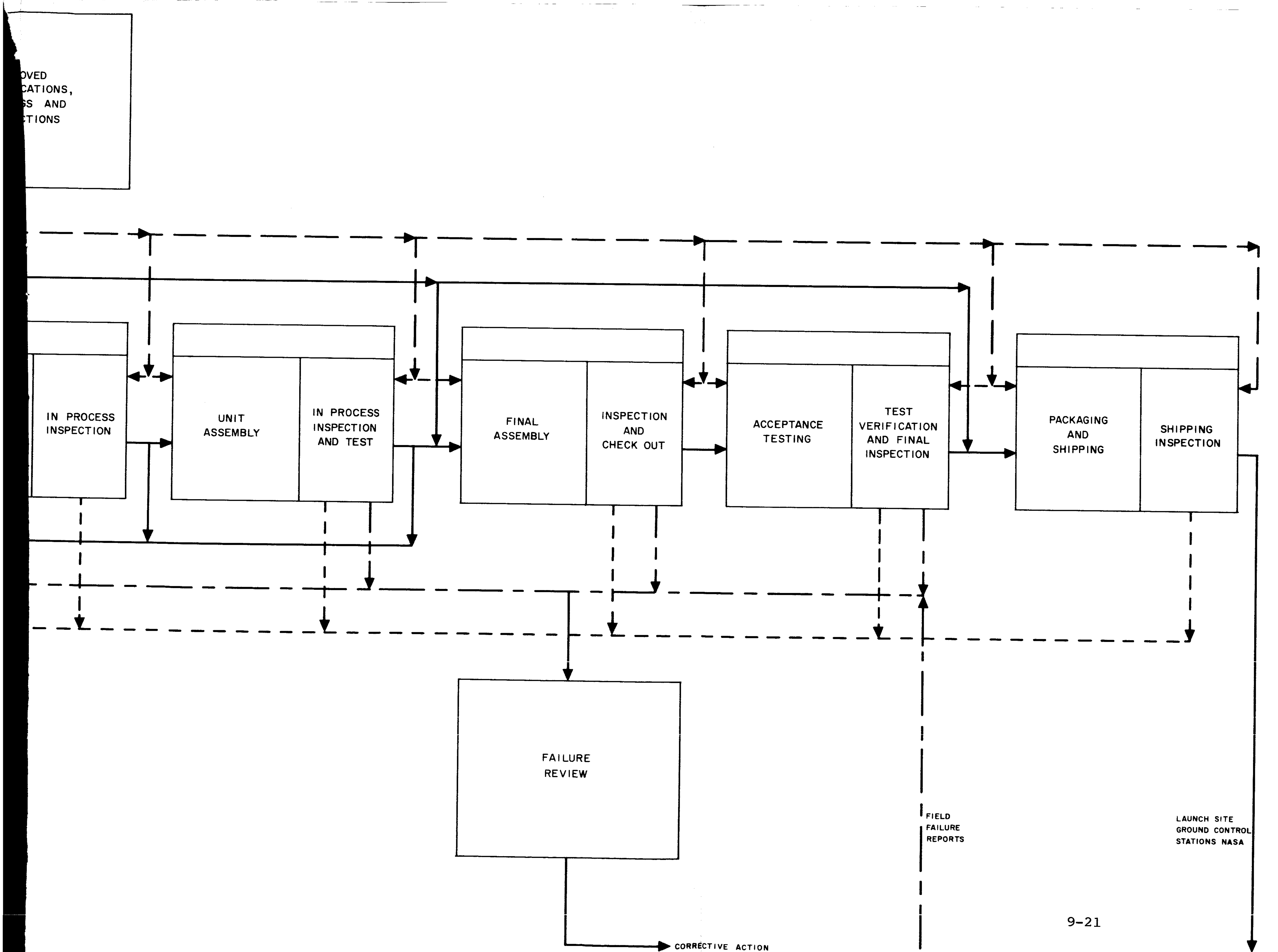


Figure 9-1. General Flow of Material including Control Points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CATIONS,
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CTIONS
IN PROCESS INSPECTION
UNIT ASSEMBLY
IN PROCESS INSPECTION AND TEST
FINAL ASSEMBLY
INSPECTION AND CHECK OUT
ACCEPTANCE TESTING
TEST VERIFICATION AND FINAL INSPECTION
PACKAGING AND SHIPPING
SHIPPING INSPECTION
FAILURE REVIEW
FIELD FAILURE REPORTS
LAUNCH SITE GROUND CONTROL STATIONS NASA
CORRECTIVE ACTION

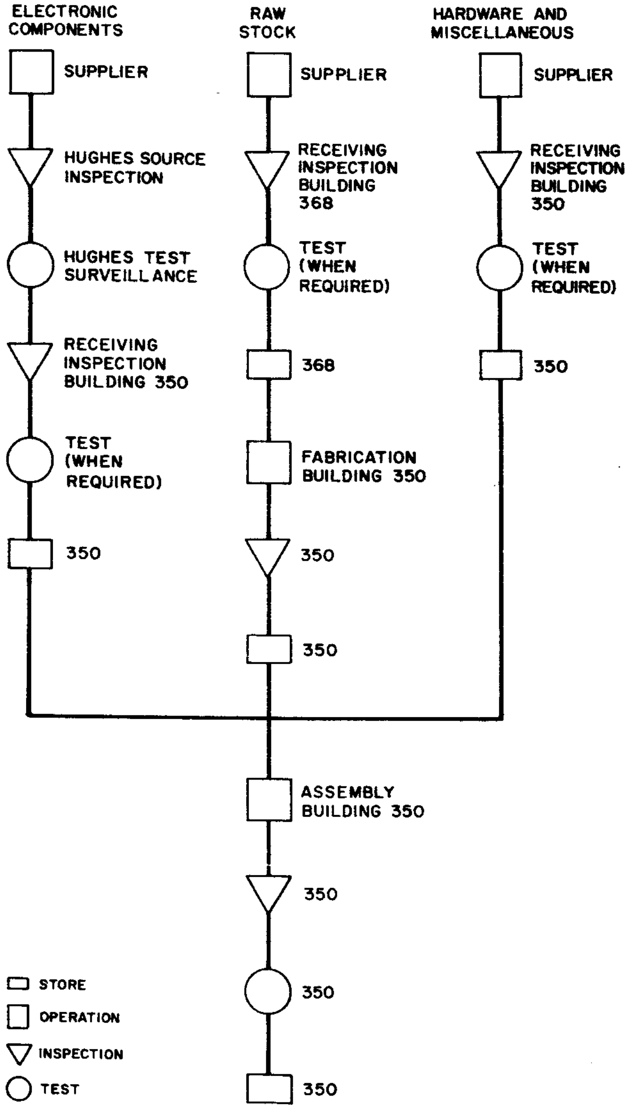

Figure 9-2. Flow Chart for Control Items Listed

Power subsystem, structure, harness, sun sensor, switch

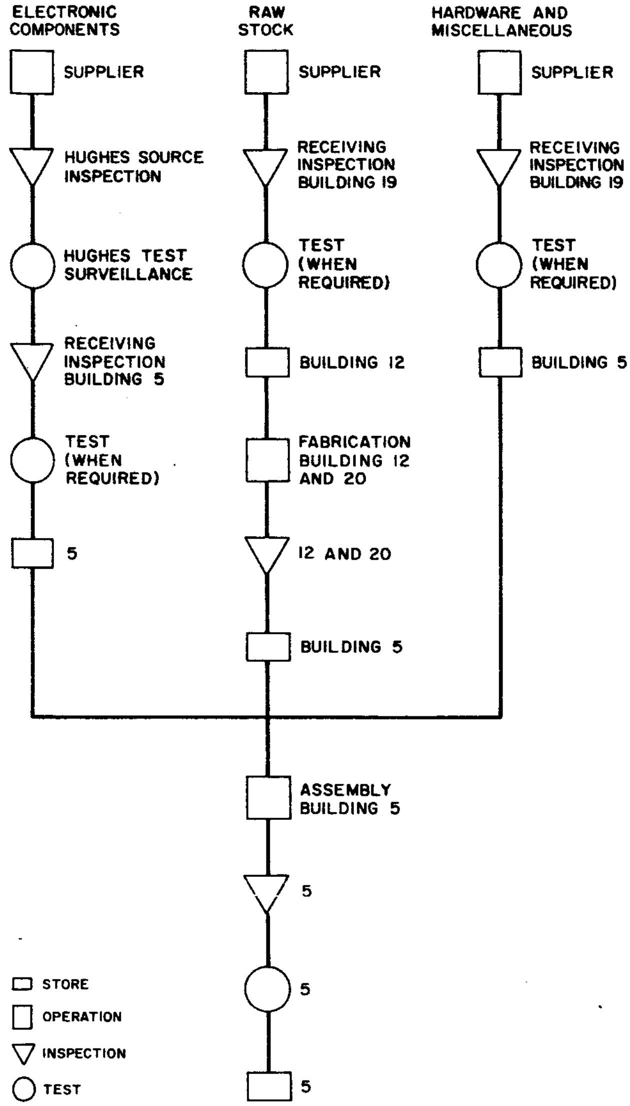

Figure 9-3. Flow Chart for Control Items Listed

Transponders, transmitter, communications antenna, phased array control unit, telemetry and command antenna system, command subsystem, telemetry subsystem

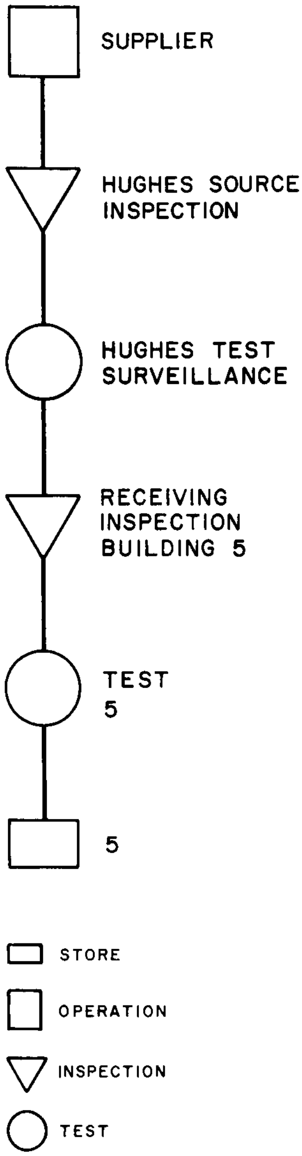

Figure 9-4. Traveling-Wave Tube Flow Chart

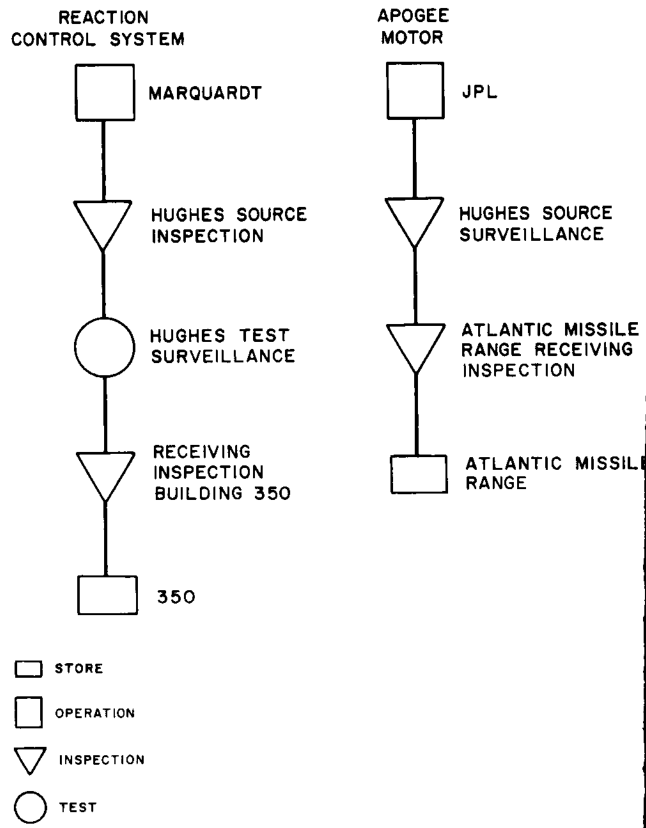

Figure 9-5. Flow Chart

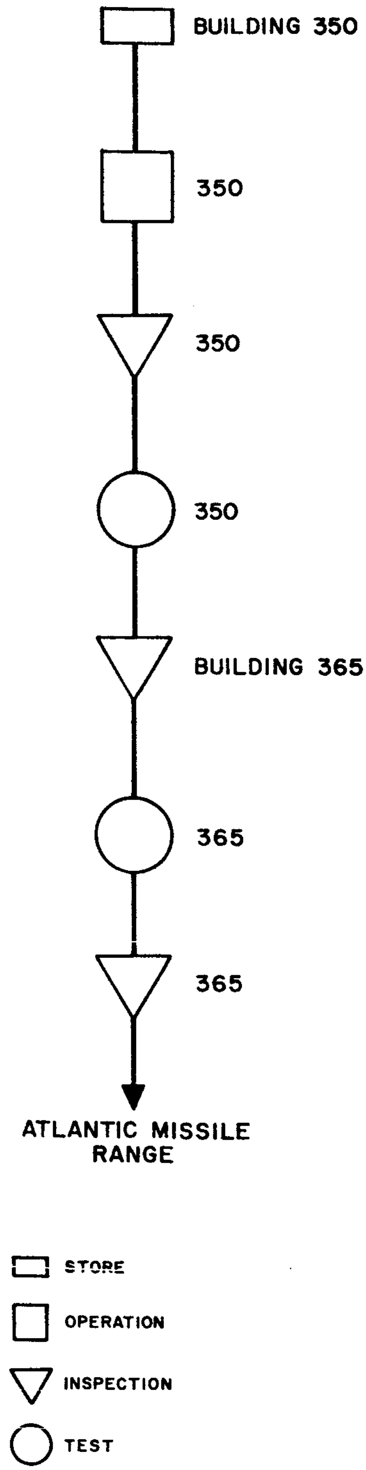

Figure 9-6. Spacecraft Assembly Flow Chart

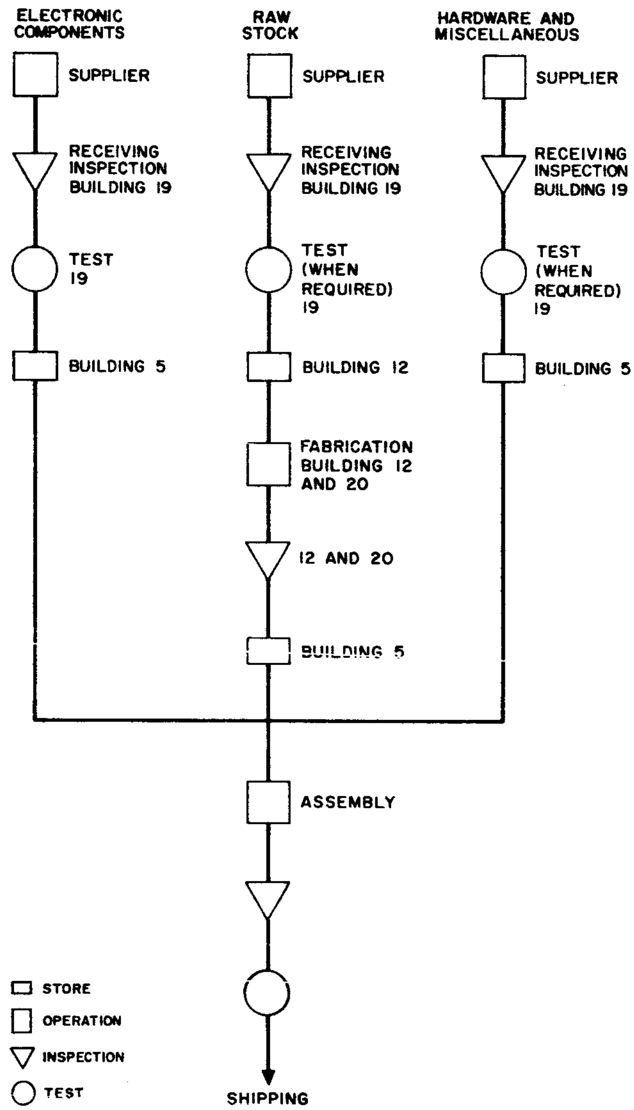

Figure 9-7. Flow Chart for Ground Control, Test, and Special Handling Equipment (Culver City)

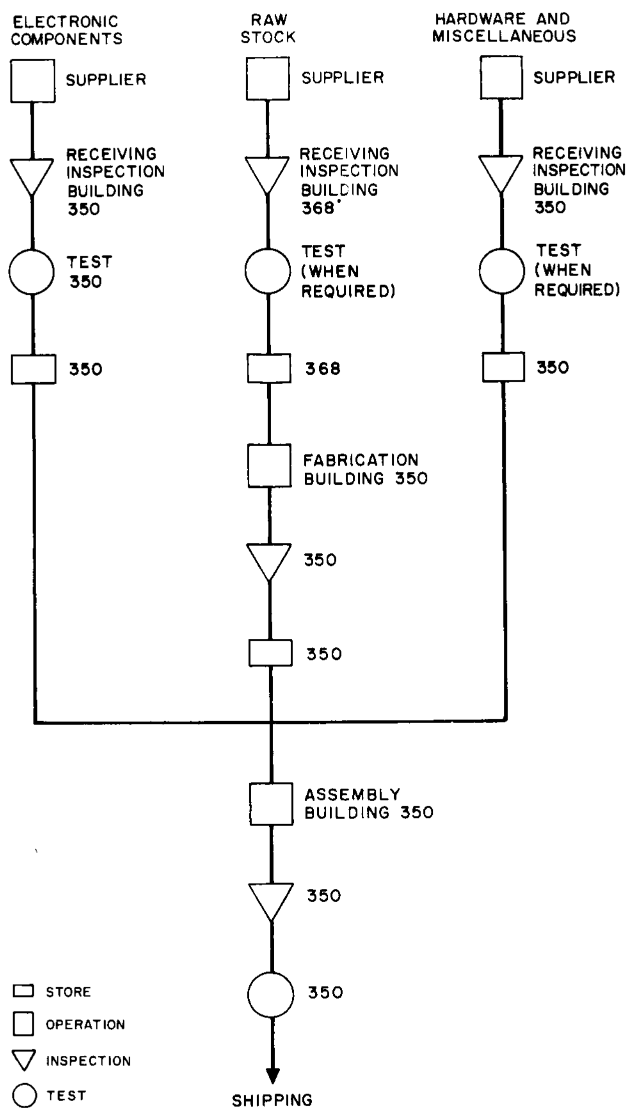

Figure 9-8. Flow Chart for Ground Control, Test, and Special Handling Equipment (El Segundo)

7.0 Milestones

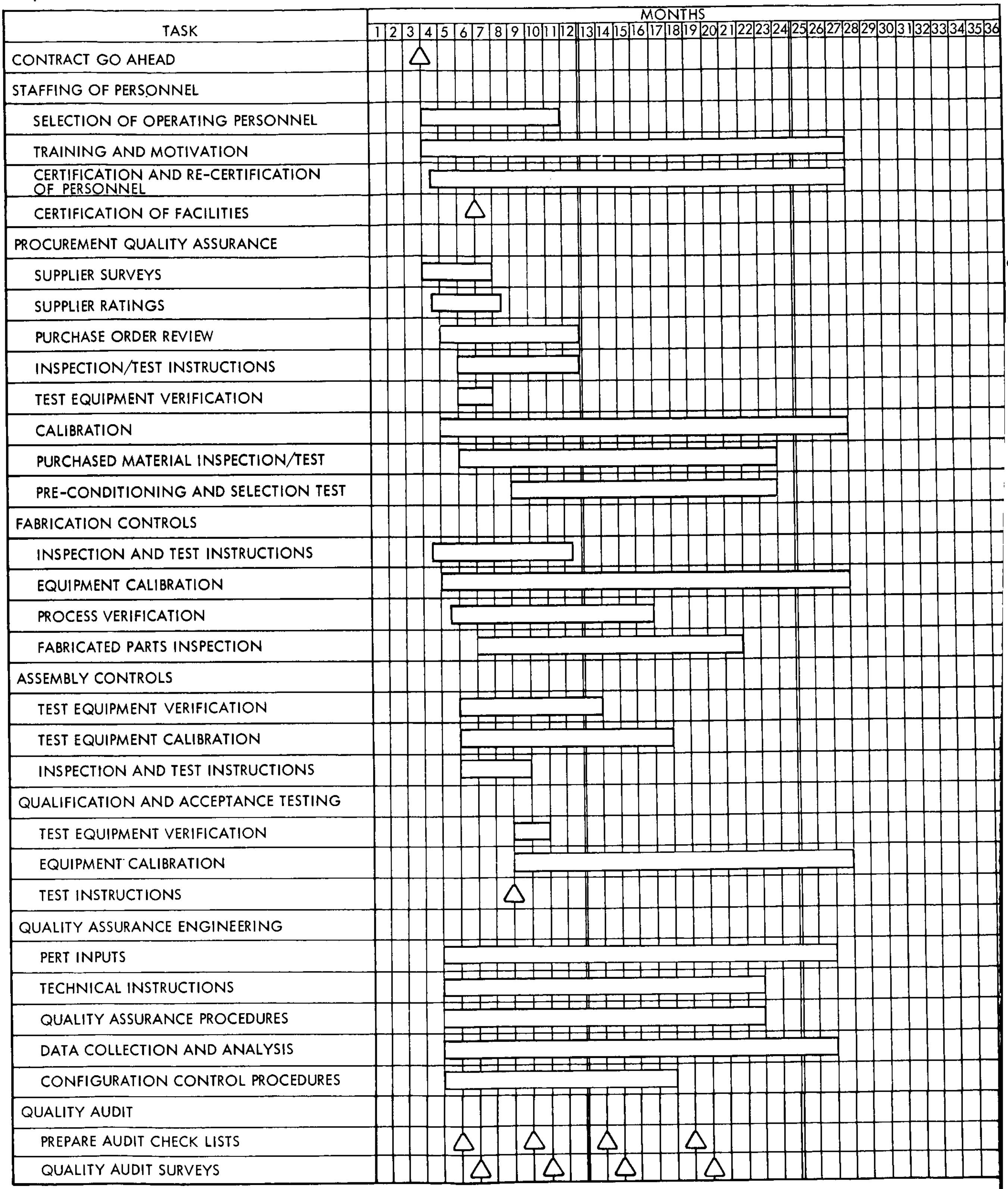

Figure 9-9. Quality Control Operating Plan
Milestone schedule